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Implementation of Rotorcraft Damage Tolerance (RCDT): Technical Issues, Challenges, and Approaches

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ABSTRACT

This document presents interim results of research efforts aimed at providing a resource on the application of fracture mechanics based damage tolerance to metallic rotorcraft structure for engineers involved in the design, analysis, certification, and maintenance of rotorcraft structure. This report focuses on the technical issues and challenges associated with the implementation of rotorcraft damage tolerance (RCDT) for metallic rotorcraft structure. Related issues such as those associated with fleet support (logistics, procurement, etc.) are not addressed. While a fairly comprehensive listing of technical issues is included, not all of the issues are explored in this interim version of the report. Technical issues addressed to some extent in this report include usage spectra, load spectra, stress intensity analysis, material crack growth properties, crack growth analysis, and validation. It is expected that future updates of this document will expand on the above topics and will also address other issues such as the effects of fatigue life enhancement processes on crack growth, usage monitoring, initial crack size, NDI techniques and capabilities, and risk assessment/reliability.

The preparation of this document has been a collaborative effort between Bell Helicopter Textron Incorporated, The Boeing Company, and Sikorsky Aircraft Corporation and was produced under an on-going NRTC/RITA project per NASA Cooperative Agreement # NCCW-0076 Entitled "Advanced Rotorcraft Technology". Consistent with this cooperative agreement, funding for this project is shared equally between industry (Bell, Boeing, Sikorsky) and government (FAA). It should be noted that in addition to this project, the FAA is funding numerous other research projects related to RCDT.

1. Introduction

1.1 Document Description/Disclaimers

The intent of this document is to provide a resource on the application of fracture mechanics based damage tolerance to metallic rotorcraft structure for engineers involved in the design, analysis, certification, and maintenance of rotorcraft structure. This report focuses on the technical issues and challenges associated with the implementation of rotorcraft damage tolerance (RCDT) for metallic rotorcraft structure. Related issues such as those associated with fleet support (logistics, procurement, etc.) are not addressed.

The preparation of this document has been a collaborative effort between Bell Helicopter Textron Incorporated, The Boeing Company, and Sikorsky Aircraft Corporation and was produced under an on-going NRTC/RITA project per NASA Cooperative Agreement # NCCW-0076 Entitled "Advanced Rotorcraft Technology". Consistent with this cooperative agreement, funding for this project is shared equally between industry (Bell, Boeing, Sikorsky) and government (FAA). It should be noted that in addition to this project, the FAA is funding numerous other research projects related to RCDT.

This Interim version of the document represents an initial effort to combine all of the technical aspects associated with the implementation of RCDT. Future updates to this document are planned for subsequent years in the on-going project. The preparation of the current document was focused on identifying technical issues and challenges, presenting related results from work accomplished under the RCDT project, and presenting plans of pertinent future work under the RCDT project. The inclusion of related research from literature, prior NRTC/RITA projects, or other sources has been limited for this initial version of the document due to time and budget constraints. It is expected that future versions of the document will include this material along with updates relating to progress on the RCDT project.

Comments, suggestions, and recommendations from readers are encouraged. RCDT is a complex subject that encompasses many areas of expertise. The preparers in no way claim cognizance of all available relevant information. Any help that readers may wish to provide can be addressed to any one of the individuals listed below:

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While every effort has been made to assure that data and conclusions presented herein are accurate and unbiased, users should be aware that these efforts might not have been completely successful. The collaborators in this document (Bell, Boeing, and Sikorsky) accept no liability for the use or mis-use of the information contained herein nor should their participation in the preparation of this document be taken as an endorsement by any of the three participating companies of some or all of the data, processes, procedures or methods related herein. Users should recognize that the application of RCDT is an evolving technical area and that some of the information presented herein could become obsolete over time as advances are made in various research areas.

1.2 Background

The safe and economic management of fatigue loaded structure is a primary consideration in rotorcraft. The very nature of rotary wing flight results in fatigue loads of various types (axial, bending, shear, torsion) and characteristics (load frequencies, load ratios, etc.) in different sections of rotorcraft structure. Rotorcraft structure includes dynamic components (moving components such as blades, rotor heads, rotating system controls, drive system, fixed system controls) and stationary components (airframe, transmission housings, landing gear, etc.). Due to the efficiencies required to achieve economical and useful rotary wing flight, many of these components are both single load path and flight critical. While many different materials are utilized in rotorcraft structure (metals, composite materials, high temperature plastics, etc.) metals are a primary choice, in particular for many dynamic components that are single load path, flight critical structure.

The management of fatigue critical metallic components is primarily concerned with cracks that can develop under fatigue loading and can lead to the loss of function. The phenomenon of fatigue cracking in metal components can be thought of as occurring in two stages. The first stage is the fatigue crack initiation or micro crack growth stage where the accumulation of fatigue "damage" over time leads to the development of a small but measurable macro crack. When

small, these fatigue cracks do not necessarily compromise the functionality of a component. The second stage is where an existing fatigue crack grows under fatigue loading until the crack reaches a critical size at which the structural function of the component is compromised.

Historically, rotorcraft fatigue critical metallic components have been managed by employing the "safe life" approach. The primary objective of the safe life approach is to establish a component retirement time such that the component is retired from service while the risk of the "initiation" of a fatigue crack is still very low. The basic elements of the safe life approach methodology as applied by most US rotorcraft manufacturers consist of aircraft usage (occurrence rates for flight regimes and GAG cycles), flight loads corresponding to the regimes in the usage, component fatigue strength, and Miner's Rule for the linear accumulation of fatigue damage. While the basic elements are common within the industry, some of the details of the procedures followed within each element can vary between companies.

Typically aircraft usage is established based on FAA or Military standard requirements plus consideration of expected or actual usage from operators. The philosophy behind the definition of usage occurrence rates for the various regimes is to include a conservative bias (higher than average occurrence rates) for regimes with high fatigue loads to provide coverage for aircraft that might experience more extreme usage than the average aircraft within the covered "fleet". The occurrence rates for the developed usage are generally expressed on the basis of a specific number of flight hours. For example, some companies choose to define assumed usage and the corresponding regime occurrence rates on the basis of 100 hours of flight. Others are known to use 1 hour of flight as the basis for defining occurrence rates.

Because of the complexity of the loads and load paths in rotorcraft, the flight loads for the regimes defined in the aircraft usage are usually based on loads measured during actual flight. The flight strain surveys used to measure these loads are usually designed to provide conservative or "above average" loads. This can be accomplished by measuring the loads for each regime at some extreme range of the regime envelope (such as extreme gross weight, extreme center of gravity, extreme "g" level for the regime "g" level range, etc.). Often this approach is combined with multiple repeats so that more than one load sample is available for each regime. For fatigue life calculations the worst-case sample, or Top of Scatter (TOS) load sample, is often used as the criteria for selecting the load sample to be used for the fatigue damage calculations for that regime.

Component fatigue strength is typically established by fatigue testing full-scale components. Often these tests are conducted on assemblies of components per aircraft specifications to assure duplication of response to flight loading

conditions when the load paths and/or the phasing of loads from different sources are complex. In this application, fatigue strength is usually taken to be the development of a macro fatigue crack. That is, when the presence of a crack is detected the testing is stopped and the datum point is taken to be a fatigue failure. A common practice is to test multiple specimens (usually six). A mean SN (Stress or load versus cycles to failure) curve is established from this test data. For fatigue damage and life calculations, this mean curve undergoes a reduction to create a "working" curve. The most common reduction is a three standard deviation reduction from the mean curve that results in the so-called "Mean minus three sigma" (M-3σ) SN curve. The fatigue tests are usually conducted in terms of the same load parameters that are measured during the collection of flight loads. As a result, a precise stress analysis at critical sections is not required. The fatigue testing also helps determine the critical failure sections for a component. For some components with different sections that have comparable margins, there can be multiple critical sections, for instance for components that are affected by multiple load paths.

Miner's Rule for the linear accumulation of fatigue damage is typically used to determine the calculated replacement time for a given component. The first step in the calculation process is to prorate the number of load cycles for each regime to be consistent with the occurrence rates in the assumed usage. These loads are then used in conjunction with the working fatigue strength to calculate the fatigue damage for each load cycle. The total calculated fatigue damage then represents the amount of fatigue damage accumulated over the time basis for which the usage is defined. The ratio of the time basis for the usage to the damage per that time basis is then typically considered the calculated replacement time.

The safe life approach has been successfully employed in the rotorcraft industry for many years. One criticism of this approach is that it typically does not directly address some factors that can influence the fatigue life of a component, such as corrosion, handling damage, non-conforming parts that elude quality checks, etc. If one adopts the following definitions of the different categories of rotorcraft fatigue (adapted from Reference 1), namely:

- *Normal fatigue* - Nominal condition parts, usage within expected ranges, loads consistent with measured loads,
- *Unexpected normal fatigue* - Nominal part with unanticipated usage or loads, and
- *Anomalous fatigue* - Unpredictable due to such occurrences as material degradation due to manufacturing or service related defects, scratches, dings, corrosion, etc.,

then the safe life approach most directly addresses the normal fatigue case. While the conservatisms built into the safe life approach could provide some

margin for unexpected fatigue or anomalous fatigue, there is no way to assess the extent of this coverage.

Unexpected normal fatigue represents the case where the safe life approach does not succeed because the assumptions and data used for the safe life methodology do not represent the actual experience of the component. The success of other prognostic approaches, such as fracture mechanics based damage tolerance, would also be affected by this lack of accurate input information. One way to address unexpected fatigue for prognostic approaches such as safe life or fracture mechanics based damage tolerance is through monitoring aircraft usage or actual aircraft loads. Much progress has been made in the rotorcraft industry in developing the needed technologies to implement individual aircraft monitoring. Another way to address unexpected fatigue is to use a diagnostic approach where various techniques would be used to detect the presence of a crack or crack like flaw. While some progress has been made in this area for rotorcraft applications, much development work still needs to be done before large-scale implementation of such systems are feasible.

One approach for addressing anomalous fatigue is a modified version of the safe life approach that is sometimes referred to as flaw tolerance. In this approach, the fatigue testing is conducted on components that feature intentionally introduced damage in critical locations that is intended to simulate manufacturing, handling, or in-service damage such as scratches, dings, and corrosion. This damage tends to act as crack initiation sites and tend to develop macro fatigue cracks at a lower number of cycles than would occur in pristine specimens. As a result, the working fatigue strength tends to be lower and calculated replacement time is also lower. Some of the drawbacks of this approach include the difficulty in characterizing the many variations of damage that can be introduced during manufacturing or service and the inability to account for subsurface defects that can originate during manufacturing.

Another approach for addressing anomalous fatigue is by a fracture mechanics based damage tolerance approach. In this approach, the presence of a small fatigue crack is assumed. As a result, the full characterization of the mitigating factors that could have influenced the initiation of the crack, such as environmental, manufacturing, or handling damage have less significance than for the flaw tolerance approach.

The core process in the fracture mechanics based damage tolerance approach is the calculation of the growth of an initial fatigue crack that is assumed to exist in a component. This requires information such as the location and orientation of the crack in the component, the definition of the stress spectra and corresponding "spectra" in terms of the stress intensity factor (a key fracture mechanics parameter), and the fracture mechanics characteristics of the

material. While the steps required in this core process are fairly straightforward, in practice there are many complex considerations, issues, and technical challenges associated with a large scale implementation of a fracture mechanics based approach for managing fatigue critical rotorcraft structure. The following chapters of this report identify and discuss many of these issues and challenges and present some approaches for addressing these concerns.

2. Rotorcraft Damage Tolerance (RCDT) Technical Issues and Challenges

Damage tolerance analysis, or more specifically as pertains to this program, crack growth analysis in metallic materials, has been successfully applied in many industries (nuclear, fixed wing aircraft, aircraft engines) to assure safety of structures. Up to the current date, the application of damage tolerance analysis to rotorcraft structures has been limited due to some unique challenges presented by rotorcraft structures.

As applied to rotorcraft, crack growth based damage tolerance presents issues and challenges that must be addressed in order to establish the needed technical foundation and efficiencies. Reference 2 presents a discussion of many of these issues and also presents a "road map" of needed research. The road map identifies ten areas of needed research as:

Table 2-1 Summary of Rotorcraft Damage Tolerance (RCDT) Research Road Map Areas (Reference 2).

Road Map Area	Description
1	Rotorcraft Damage Tolerance (RCDT) specific issues study
2	Spectrum development and usage monitoring
3	Equivalent initial flaw (crack) sizes (EIFS)
4	Crack growth database
5	Nondestructive inspection/evaluation (NDI/E)
6	Certification Testing
7	Life enhancement methods
8	Crack growth Analysis (CGA)
9	Risk assessment methods
10	Corrosion control

A convenient way to present and discuss the issues and challenges associated with the implementation of RCDT is within the context of the overall RCDT methodology. The basic elements of a damage tolerance analysis methodology for rotorcraft applications are presented in Figure 2-1. The figure includes references to the road map areas of Table 2-1 in parentheses. The core of the methodology is the capability to establish safe operational inspection intervals or retirement times by conducting crack growth analyses at known levels of risk. A requirement is that this process has a strong technical foundation. In addition, in a production environment this process must be efficient.

As applied to rotorcraft, the elements in Figure 2-1 present issues and challenges that must be addressed in order to establish the needed technical foundation and efficiencies. These issues are summarized in Table 2-2 and are discussed in the following sections.
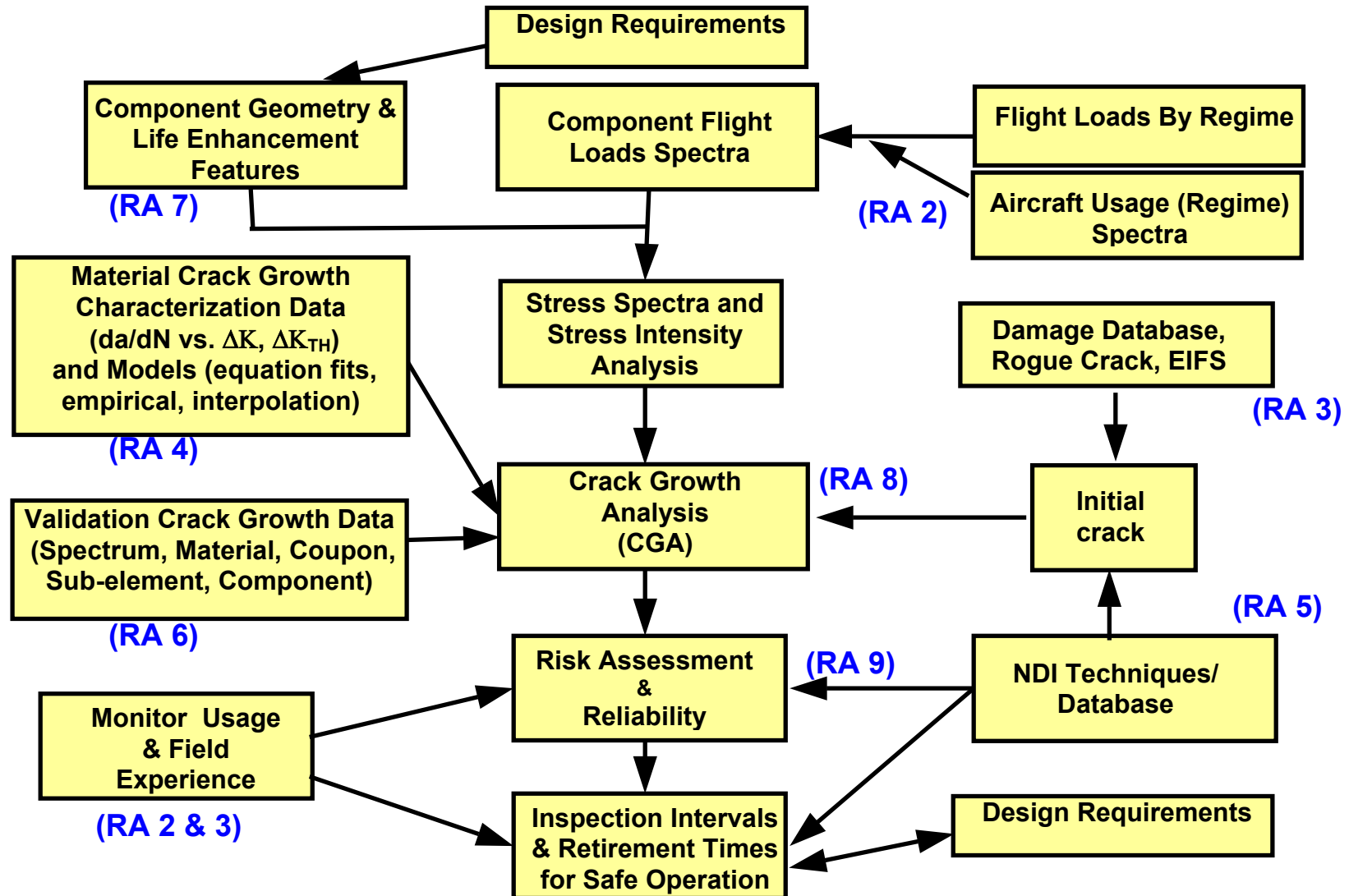
Figure 2-1 Basic Elements of Rotorcraft Damage Tolerance (RCDT) Methodology.

Table 2-2 Rotorcraft damage tolerance issues and challanges.

ISSUE	COMMENTS
Load Spectra Issues	Rotorcraft load spectra are developed from usage and the loads associated with that usage
Usage Variability (Flight regime occurrences/sequences)	Usage can vary greatly within a fleet due to the diverse missions which rotorcraft perform
Loads Variability between repeat occurrences of a given flight regime	Fatigue loads that occur during a given flight regime will exhibit some variability for each occurrence of that flight regime.
Load Characteristic Variability	A variety of fatigue loads with different characteristics are generated throughout rotorcraft during operation. Some of the important different characteristics include frequency content (high frequency dominant, low frequency dominant, mix of high and low frequency) and load ratio (single dominant load ratio, wide range of load ratios, negative load ratios versus positive load ratios)
Load Spectra Variability	As a result of usage variability and loads variability, the load spectra for a given fatigue load parameter has variability
Loads Processing	For any fatigue analysis (safe life or crack growth), time history loads must be converted into steady and alternating load cycles. Different approaches can be used, with potentially different results.
Loads Selection Criteria	For the safe life approach, where the linear accumulation of fatigue damage is assumed, conventions such as using top of scatter loads and biasing occurrence rates for high load (and thus high fatigue damaging) regimes imposes a level of conservatism in the calculation. For crack growth analysis if non-linear effects are modeled, these conventions for selecting and weighting loads could lead to an un-conservative result.
Efficient Load Spectra Development	As a result of the above, the development of load spectra is complex; tools and processes are needed to accomplish this efficiently
Implementation of Usage or Loads Monitoring	For the implementation of usage or loads monitoring as a means for applying a variable inspection interval or retirement life, a reliable and efficient means for collecting monitored data and updating crack growth calculations is needed.
Geometry Issues	Rotorcraft components can have complex load paths and complex geometry at critical sections.
Stress Spectra Development	Stress spectra development for a given load spectra can be complex for complex load paths and part geometry
Stress Intensity Factor Determination	Stress intensity factor development depends on geometry, crack dimensions, and stress state. Many solutions are available for common simple geometries; however, for complex geometries and load paths numerical solutions are needed. Also, as the crack grows, the presence of the crack could affect the stress distribution.
Crack Growth Path	Local stress spectra and stress intensity factors vary as cracks grow. Properly accounting for these variations is challenging for complex geometry and load paths. Also, in certain cases such as with complex geometries the crack growth direction is not obvious; criterion, such as those that use strain energy density, is needed to determine the direction of crack growth.

Table 2-2 Rotorcraft damage tolerance issues and challanges. (continued)

ISSUE	COMMENTS
Life Enhancement Issues	Processes used to improve fatigue strength, such as shot peening, carburized surfaces, and cold working add complexity to crack growth analysis (stress spectra, stress intensity, material crack growth properties)
Material Crack Growth Property Issues	Most crack growth property data available today has a poorly defined threshold region and was developed using thin coupon specimens with through cracks
Threshold data	Accurate definition of threshold region is required to address the high cycle fatigue elements of rotorcraft load spectra; questions have been raised over whether current established test methods adequately address plasticity effects.
Stage II (stable crack growth) data	Properly modeling stress ratio effects can be very important for some rotorcraft fatigue load parameters.
Fracture Toughness	Required but not so critical for most rotorcraft applications where the high frequency of loading generally results in very rapid crack growth before maximum stress intensity values come close to the fracture toughness.
Environmental Effects	Temperature, atmospheric conditions (humidity, salt), other (hot oil, etc.), loading frequency, etc.
Design Issues	
Design Requirements	Design requirements for some components (weight, single load path, physical envelope) can result in components where damage tolerance analysis is not the best approach
Initial Crack Size Issues	The initial crack size used for crack growth analysis must consider both the capability of applicable inspection techniques and the equivalent initial flaw size for "damage" which could be incurred on components.
Inspection Techniques	Numerous inspection techniques are available depending on many factors including material, geometry, the inspection environment (field inspection, overhaul facility inspection), etc. The risk associated with the assumption of an initial crack size is dependent on many factors (technique, inspector capability, inspection environment) and is difficult to quantify.
Equivalent Initial Crack Size for "damaged" components	

Table 2-2 Rotorcraft damage tolerance issues and challanges. (continued)

ISSUE	COMMENTS
Crack Growth Analysis Issues	
Modeling of crack growth characteristics (da/dn vs. ΔK, ΔK_{TH}, stress ratio effects)	The method used to represent the characteristic crack growth data can affect results.
Load Interaction Effects	Rotorcraft fatigue loads can consist of complex load spectra with a wide range of load ratios and maximum and minimum loads for the different load cycles. This can possibly introduce significant load interaction effects that can vary with material.
Crack Growth Path	Proper modeling of crack path in complex geometry sections is required to properly account for any stress redistribution and corresponding effect on the stress intensity factor
Validation	Crack analysis process capability to properly account for complex load spectra (interaction effects on the diverse materials used in rotorcraft) and complex geometry must be validated
Efficiency and Ease of Use	
Certification Issues	Demonstrate that approaches used for loads spectra, crack growth properties, and crack growth analysis are acceptable and consistent with risk assessment methods and initial crack size assumptions.
High Cost of Full Scale Component Testing	Full scale component crack growth testing is much more complex than full scale component testing for safe life applications (crack initiation); this results in much higher costs and much longer test times.
Crack Growth Analysis Processes	Demonstrate that the crack growth analysis processes for the assumed usage/loads/initial flaw size are valid.
Overall Methodology	Demonstrate that the assumptions made in crack growth analysis input (usage/loads/initial flaw size) are valid within the context of the overall methodology.
Inspection Issues	
Inspection Environment	Field inspection capabilities are limited; Depot level inspections allow for more capability but can be more costly.
Risk Assessment/Reliability Issues	All of the above affect Risk Assessment/Reliability
Inherent scatter of crack growth	Crack growth will exhibit some variability for repeated tests under the same conditions (loading, specimen material and geometry, test environment, etc.)

2.1 Load Spectra

Typically the first step in developing a load spectrum is to define a usage spectrum that defines occurrence rates for flight regimes (level flight, turns, etc.) and flight related low cycle conditions (ground air ground cycle type loads). A load spectrum is then developed for each fatigue load parameter by assigning appropriate loads for each regime and condition. Rotorcraft structures experience a diverse range of fatigue load spectra due to many factors. For a given fatigue load parameter the diversity of usage (flight regimes associated with diverse missions) that rotorcraft can perform, the variability of loads between each occurrence of a flight regime or mission, the criteria for selecting loads where multiple load samples are available, and the processing techniques used to develop the fatigue load cycle data from the time history loads can influence the calculated crack growth results. All of these factors will impact the eventual risk assessment of the overall methodology. In addition, rotorcraft structures experience many different fatigue loads throughout the structure with diverse characteristics that can require different capabilities with regards to crack growth analyses.

Usage monitoring and loads monitoring are emerging technologies that could potentially allow for the management of fatigue loaded structure based on the actual usage and/or fatigue load exposure to which the structure has been subjected instead of using an assumed usage. These technologies have the potential for both increased economic utilization of components and improved safety. In order to implement these technologies, acceptable means for assuring the quality of the identified usage or measured loads is needed. In addition, for these technologies to be practical, an efficient means for developing load spectra and performing crack growth analysis is needed because these steps would be repeated on a regular basis.

2.1.1 Usage Variability

Usage can vary greatly within a fleet due to the diverse missions which rotorcraft perform. Also, the relative severity of a given mission can vary between the various fatigue loads that occur throughout rotorcraft structure because of the different characteristics of rotorcraft fatigue loads and the different sources of excitation and other causes of high magnitude fatigue loads. For example, for missions that include a high rate of external cargo pick up and drop off cycles, power related parameters such as drive torque might experience more relatively high load magnitudes due to power variations relative to a standard mission but load parameters that are sensitive to airspeed and are dominated by high frequency content might experience lower loads. The sensitivities of crack growth to these variations should be understood and should be included in risk assessment or reliability evaluations of the overall methodology. In some

instances, it might be appropriate to define several usage spectra where the version used in crack growth analysis depends on the load parameter.

Another consideration when developing usage spectra for crack growth analysis is the inclusion of or exclusion of rarely occurring high load regimes. These extreme, but rarely occurring regimes are often included in safe life methodology calculations because in the application of Miner's Rule the inclusion of these loads is always conservative. However, for crack growth analysis if the non-linear effects such as crack closure are included in the calculation, the inclusion of these high loads could retard crack growth and thus would be unconservative for the large percentage of time where these regimes do not occur.

2.1.2 Load Characteristic Variability

For a given rotorcraft, there are a number of different significant fatigue loads with different characteristics. Some loads are high frequency (one or more load cycle per rotor revolution) such as loads in main and tail rotor components and controls. Other loads can be of lower frequency, such as one per flight maneuver or one per flight segment. Some loads can include both high frequency and low frequency characteristics. As a result, on a given rotorcraft performing a well-defined mission, there will be diverse load spectra for different load parameters. Table 2-3 summarizes some of the different characteristics of rotorcraft fatigue loads. Examples of waveforms for a high frequency (one cycle per rotor revolution) dominant rotorcraft fatigue load and a low frequency (1 cycle per maneuver) dominant rotorcraft fatigue load are shown in Figure 2-2. The figure shows time histories for the two loads during a pull-up maneuver. The lower load is the high frequency load with a frequency of one load cycle per rotor revolution. This load parameter is a reversed bending load that exhibits a short period where the one per rotor revolution alternating load magnitude first increases significantly and then decreases. For the upper load, the high frequency content is predominantly three per rotor revolution. However, the magnitude of the high frequency alternating loads are much smaller than the one per maneuver alternating load that results from the maximum load at around 2.1 seconds that is associate with the gradual shift upward of the steady load and the minimum load at around 3.6 seconds that is associated with a gradual shift downward of the steady load. These shifts in the load correspond to the variations in power requirements during the maneuver.

Sample load spectra for the two load parameters shown in Figure 2-2 that were developed for the same usage spectrum are shown in Figure 2-3 and Figure 2-4. In many cases, stress fields in which cracks would propagate would be expected to follow similar trends to the load spectra. In such cases, the spectra in Figure 2-3 has a significant compressive component of the load with a fairly small range in load ratio whereas the spectra in Figure 2-4 is predominantly positive and exhibits a wide range of load ratios.

Table 2-3 Summary of some of the different characteristics of various fatigue loads found in rotorcraft.

Category Number	Frequency Category	Frequency Specifics	Examples*
1	HIGH FREQUENCY Rotor RPM order (n/rotor revolution)	n = 1	Rotor blade flap bending Rotor shaft bending Pitch Links
		n = # of blades	Fixed system controls
		n = other than # of blades	Some Controls
		n = RPM times gear ratio	Drive Shaft Bending (Shafting between engine and main gearbox) Gear Teeth
2	LOW FREQUENCY n per Maneuver Order	n = 1,2 (could be higher depending on maneuver)	Drive Torque Chord bending moment inboard of vertical pin (fully articulated rotor)
3	LOW FREQUENCY n per Flight or Flight Segment (GAG type)	n = 1	Drive Torque CF

* Most of these loads exhibit higher order frequency content that is of lower relative magnitude. In some instances, this higher order content could be important for crack growth.

Sample Load Time History
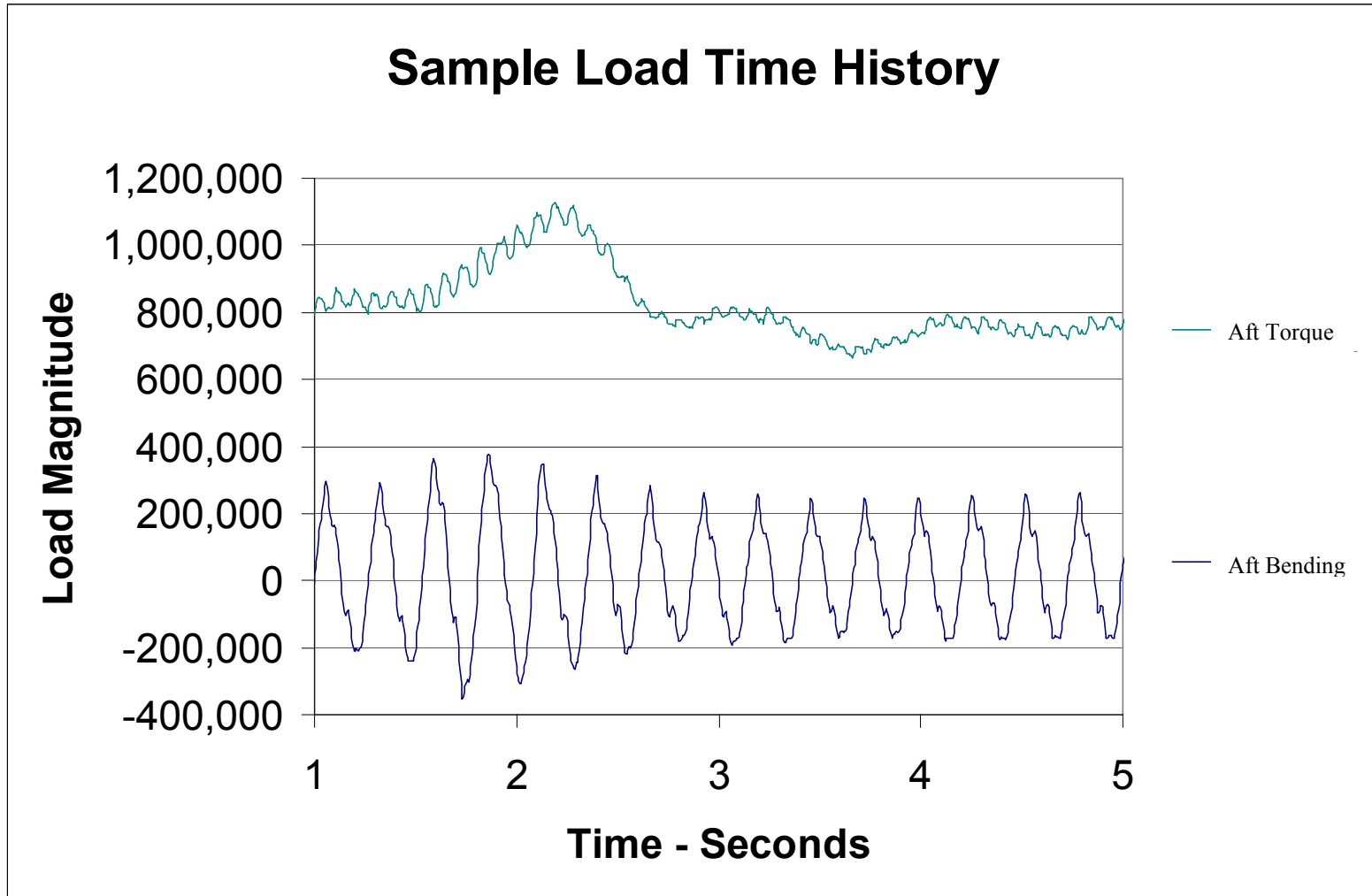
Figure 2-2 Sample Load Time Histories.

Aft rotor shaft bending, logging, 11−26−01
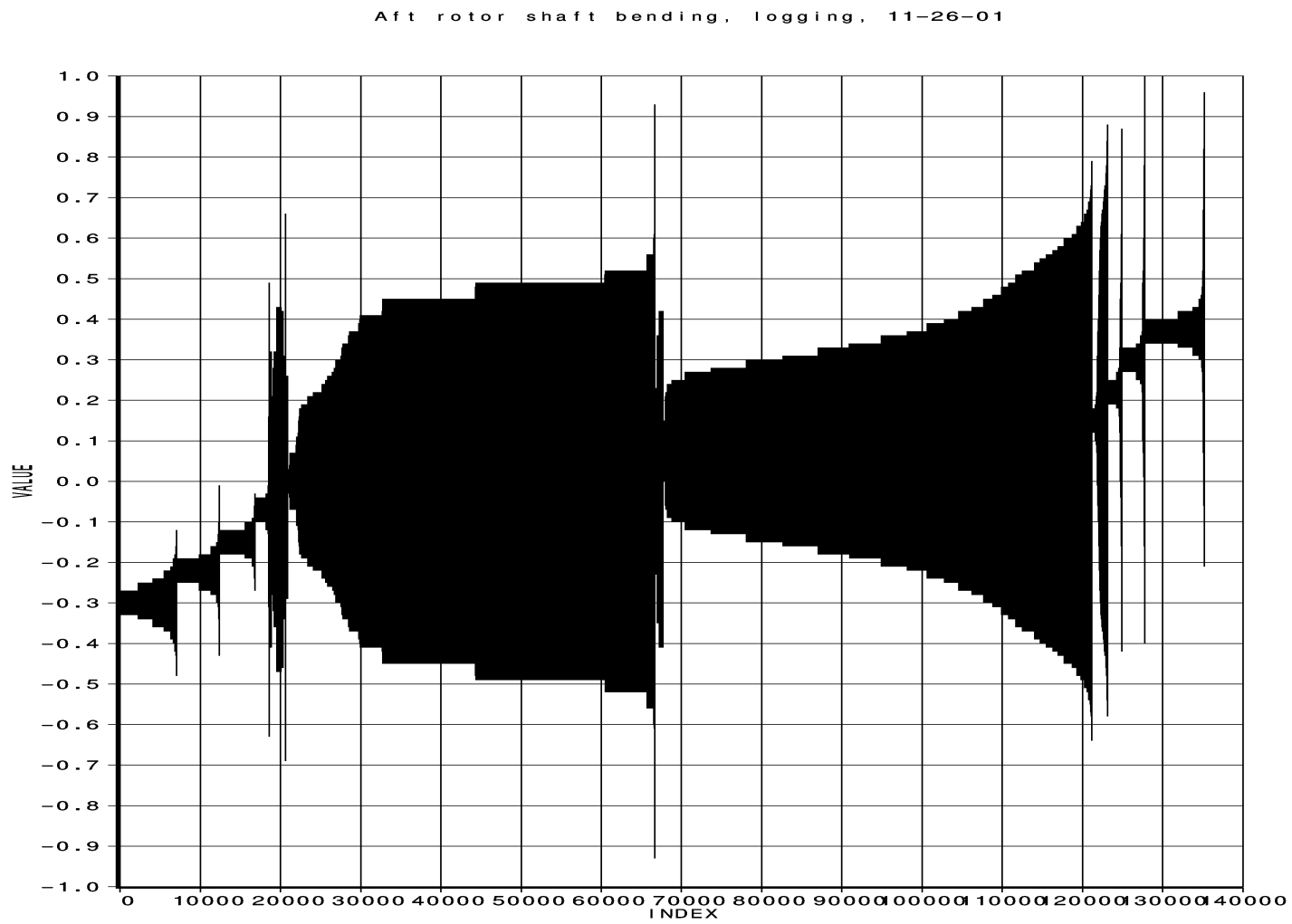
Figure 2-3 Sample normalized load histogram, one per rotor revolution dominant load.
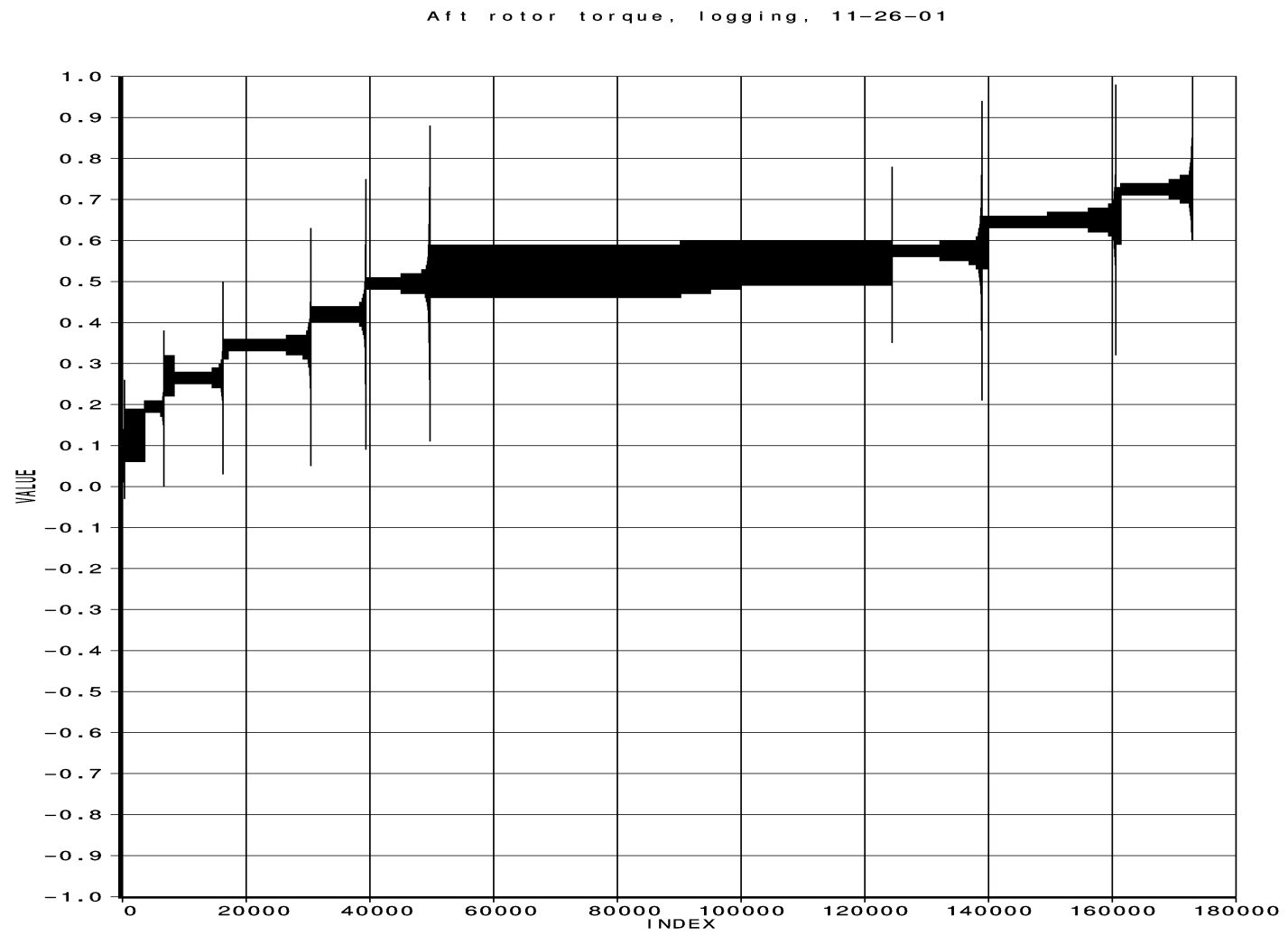
Figure 2-4 Sample normalized load histogram, low cycle dominant load.

It might be expected that for materials where crack closure is a significant factor, the spectra from Figure 2-4 might be more sensitive to these effects than the spectra from Figure 2-3. An analytical example of this is shown in Figure 2-5 and Figure 2-6. Figure 2-5 shows the results of crack growth analyses based on the load spectra from Figure 2-3. For the analyses, different scaling factors were applied to the normalized spectra. Also, analyses were performed without modeling load interaction and also with two analytical models for load interaction that are based on plastic zone size. As would be expected, Figure 2-5 shows a decrease in hours to failure with increase in scale factor. Of more interest is the comparison of the hours to failure between the analysis done with no load interaction versus those that included load interaction modeling for a given scale factor. For the load spectrum from Figure 2-3, the results shown in Figure 2-5 indicate that there is very little difference between the results. Similar analyses were conducted for the load spectrum presented in Figure 2-4. These results are shown in Figure 2-6. In this case there are significant differences between the analysis that did not include consideration of load interaction effects and the analyses that included load interaction models in the analysis.
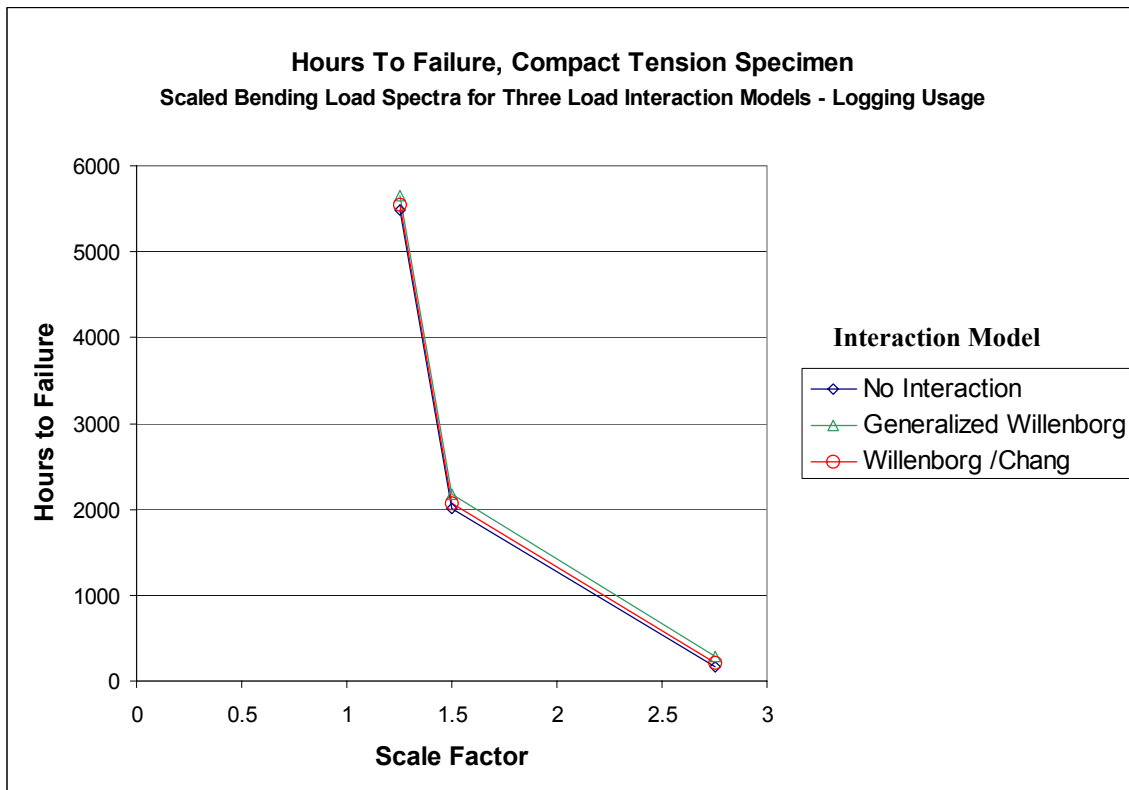
Figure 2-5 Sample crack growth analysis results for both no-load interaction and load interaction modeling using the load spectrum from Figure 2-3.
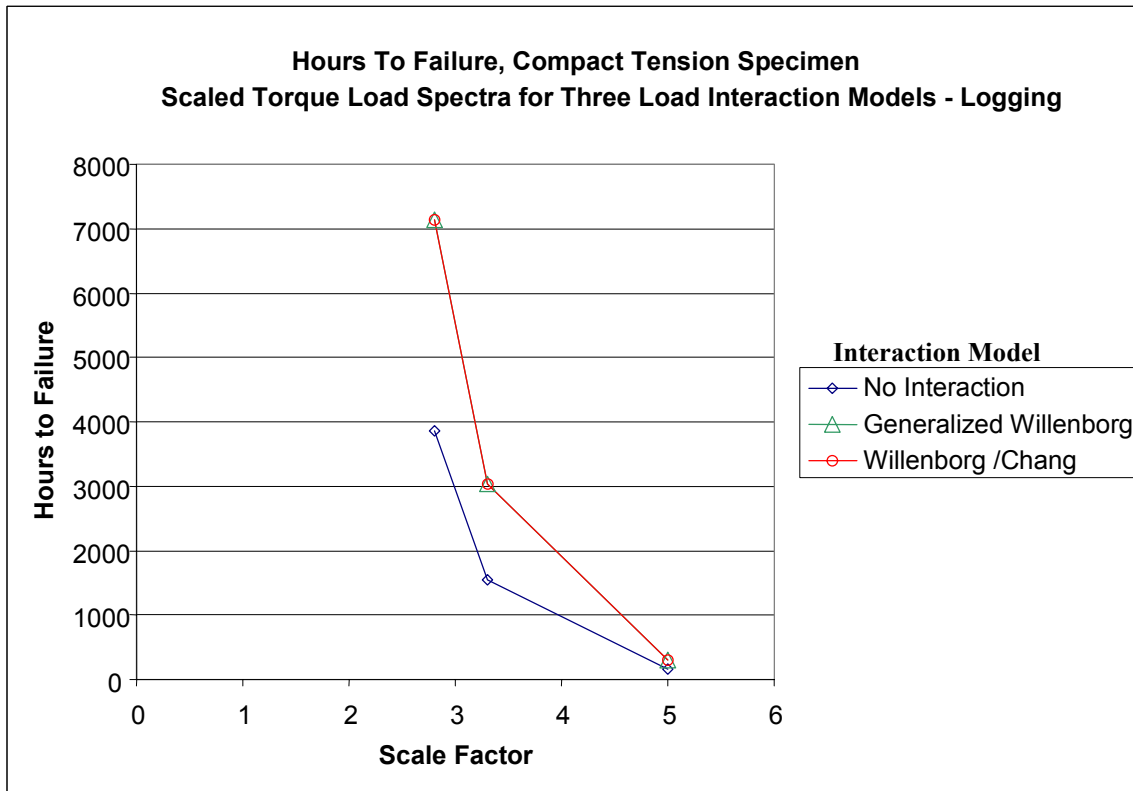
Figure 2-6 Sample crack growth analysis results for both no-load interaction and load interaction modeling using the load spectrum from Figure 2-4.

These load interaction effects could vary depending on the material. Also, analyses without accounting for load interaction effects do not necessarily always result in a conservative result relative to analyses that do include load interaction effects. This can be important when considering the risk and reliability.

One implication of these differences in the effects of load interaction is that a well devised plan that demonstrates the capability to match predicted crack growth results with testing for a given material and load spectra is not necessarily evidence that the analytical models used are adequate for predicting crack growth in a different material or for a different stress spectra. See section 3.3 for a demonstration and discussion of an approach for addressing these concerns.

2.1.3 Loads Processing

Another issue related to load spectra involves the methods by which time history flight load data is converted into steady and alternating load cycles. A common approach in rotorcraft is to use some form of cycle counting to convert the peaks and valleys from the time history data for a given load and maneuver into cycle counts for bins of steady and alternating load. This binned data is sometimes referred to as a load histogram.

The load spectra presented in Figure 2-3 and Figure 2-4 were developed from cycle counted loads data (load histograms) that were prorated to be consistent with the usage spectrum occurrence rates. The usage spectrum occurrence rates were developed to represent one hour of flight. As presented in the figures, these spectra have been arbitrarily ordered by increasing alternating load for each increase in steady load.

The use of load histograms makes it convenient to develop load spectra for usage spectra that are defined in terms of occurrence rates for a set period of time. One drawback of this approach is that the time sequencing of loads consistent with actual flight is difficult and in some instances impossible to accurately duplicate.

One approach to loads processing that would retain the time sequencing of loads is to merely reduce the time history data to a series of peak and valley values where each consecutive peak/valley pair would represent a load cycle. This approach is not very useful for applications where usage is defined in terms of a composite usage spectra because it is difficult to prorate this data into occurrence rates defined by the usage spectra and proper sequencing would also be difficult. This approach would be possible for direct load monitoring applications where actual time history loads are available for all flight time. It is also possible that this approach could be adapted to usage monitoring applications. A summary of the cycle counting approach and the sequential peak/valley approach is presented in Table 2-4.

Table 2-4 Comparisons of various approaches for processing time history loads into discrete load cycles.

Time History Load Processing Method	Strong Points	Drawbacks
Pairing of the sequence of local peaks and valleys to form load cycles (Sequential Peak-Valley)	• Maintains time sequence of loading.	• Difficult (or impossible) to combine maneuver data into an assumed usage spectrum. • Difficult to handle multiple samples of the same maneuver.
Cycle Counting over time segments on the order of a few (1-3) high frequency load cycles (high frequency cycle counting	• Easy to combine into an overall spectra for an assumed usage spectrum.	• Does not capture low cycle loads resulting from steady shifts during a maneuver. • Does not maintain time sequencing of loading.
Cycle counting over time segments for an entire maneuver (or more) where low and high cycle content is counted (Rainflow or similar)	• Relatively easy to combine into an overall spectra for an assumed usage spectrum.	• Does not maintain time sequencing of loading.

2.1.4 Usage/Load Monitoring

In the absence of usage (regime recognition) or loads monitoring, established inspection intervals and/or retirement lives are imposed on all components regardless of the aircraft and associated usage on which they are installed. These inspection intervals and/or retirement lives must provide adequate safety for components that could be exposed to the more severe usage relative to the fleet average. This imposes an economic penalty for components that are not exposed to this more severe usage. Usage or loads monitoring provides an opportunity to tailor inspection intervals and/or retirement lives for components to their actual exposure. However, the implementation of monitoring methods requires a reliable and efficient means for collecting monitoring data and an efficient process for updating crack growth analyses. This presents many

challenges, including the ability to update complex crack growth analysis such as those that require numerical methods in an efficient and timely manner. This also requires an efficient way to update load spectra as time history data in the form of usage or loads is accumulated during operation.

2.2 Geometry

Rotorcraft components frequently experience complex external loading and exhibit complex internal load paths. In addition, complex local geometry is common at critical sections. Crack growth analysis requires the evaluation of the complex stresses in crack regions and also requires the determination of the stress intensity factors. As a result, efficient tools and processes are needed to develop stress spectra and calculate stress intensity factors. Also, the crack growth path can be complex and as the crack grows the stress spectra can be affected necessitating an iterative process.

2.3 Material Crack Growth Properties

The key crack growth properties of materials that are required for crack growth analysis are based on the linear elastic fracture mechanics stress intensity factor parameter, K. This fracture mechanics parameter is a function of the local geometry at the crack tip, the square root of the crack length, and the stress. For a given geometry and crack size, the range in stress associated with a fatigue load cycle results in a corresponding range in the stress intensity factor, ΔK. A detailed discussion of this parameter is presented in section 5.2.

The key properties include the threshold stress intensity factor range (ΔK_{TH}), the crack growth rate per load cycle as a function of the stress intensity factor range associated with the load cycle (da/dn versus ΔK), and the fracture toughness (K_c). These correspond to the three regions of crack growth as described in section 5.2 and presented in Figure 5-2. The threshold stress intensity factor range corresponds to the stress intensity factor range below which crack growth will not occur (Region I in Figure 5-2). For increasing ΔK values above the threshold within Region II of Figure 5-2, the crack growth per load cycle (da/dn) increases in a stable manner. As the maximum stress intensity factor associated with a load cycle approaches the fracture toughness, the crack growth rate increases rapidly and becomes asymptotic at the fracture toughness (Region III in Figure 5-2).

Both threshold crack growth properties and stable crack growth properties vary with load ratio. Since some rotorcraft load spectra feature wide ranges of load ratios, test data to characterize material crack growth properties may be needed at multiple load ratios.

One consequence of the high frequency of loading in rotorcraft load spectra discussed above is that the crack growth time during stable crack growth (Region

II) can be very short relative to operating time (flight time). As a result, for damage tolerance analysis to be practical for some rotorcraft applications the initial crack sizes must be small and the threshold region for crack growth must be well defined. Most crack growth data that is currently available exhibits significant scatter in the threshold region. There is some evidence that this scatter could be due to plasticity effects associated with the testing technique used. Efforts to resolve the concerns with regards to the determination of ΔK_{TH} are underway under an FAA funded project at NASA Langley.

2.4 Design

Discussions of design related issues will be developed in a later version of this report.

2.5 Life Enhancement

Discussions of life enhancement related issues will be developed in a later version of this report.

2.6 Initial Crack Size

Discussions of initial crack size related issues will be developed in a later version of this report.

2.7 Crack Growth Analysis

The ability to accurately and efficiently predict the growth of cracks or crack-like-flaws in rotorcraft components and structures is of major importance in meeting damage tolerance requirements. Linear elastic fracture mechanics provides the foundation for the crack growth analysis used in this project. Details of the application of linear elastic fracture mechanics to crack growth analysis, including discussions on the stress intensity factor and material crack growth characterization are presented in Section 5.2.

The crack growth analysis process includes many steps, each of which can vary from simple to complex. A diagram of these steps and examples of levels of complexity are presented in Figure 2-7. The crack growth analysis tools that are available include "Standard" crack growth analysis codes (NASGRO, AFGROW, CRACKS2000, etc.) and Numerical crack growth analysis codes. The "Standard" crack growth analysis codes typically require stress spectra as input. Also, the "Standard" crack growth analysis codes typically provide a library of stress intensity factor solutions for common geometries. Some "Standard" crack growth codes provide for user input of tabular stress intensity factor data. These codes typically require a priori knowledge of the direction of crack growth. Numerical crack growth analysis codes typically cover the conversion of the load spectra to stress spectra, the calculation of the stress intensity factor, and the calculation of

the incremental crack growth. As a result, these codes are better suited for crack growth analysis involving components with complex geometry and load paths. These codes also typically have criteria that predict the crack growth direction as it grows incrementally.

"Standard" crack growth analysis codes can be applied to complex geometry situations if significant idealization of the geometry is acceptable. An example of such an application might be where a conservative idealization of the geometry results in a conclusion that no crack growth will occur. One drawback of this approach is that a significant weight penalty could result.

2.7.1 Modeling of Material Crack Growth Characteristics

One of the challenges in rotorcraft crack growth analysis is the analytical representation of crack growth material properties. Many rotorcraft load spectra include a wide range of load ratios. Since the threshold and stable crack growth properties of a material are dependent on load or stress ratio, the analytical representation of these properties must reflect the proper variation in properties with load ratio. In addition, test data includes scatter. The challenge then is two-fold: how to represent crack growth properties in light of the scatter in test data and how to represent crack growth properties to account for variations with load ratio.

Most crack growth analysis codes provide several options for representing crack growth material properties. Numerous equations have been developed that include a load ratio dependency. These vary in the level of sophistication. One of the more sophisticated is the NASGRO equation which provides numerous constants that can be adjusted to effect sensitivity to load ratio and curvature at the threshold region and in region III. Despite these complex equations, usually compromises have to be made where the equation matches data in some load ratio ranges very well but is somewhat off of a good fit for other load ratios. Some crack growth analysis codes provide for an empirical definition of crack growth properties (curves specified by user defined pairs of da/dn and ΔK) for different load ratios. Interpolation is then used for load ratios for which the crack growth curves are not specifically defined. This approach provides more flexibility to accurately define material property dependence on load ratio. However, it still requires judgment on how to define the da/dn vs. ΔK pairs for each load ratio within the scatter of the test data.
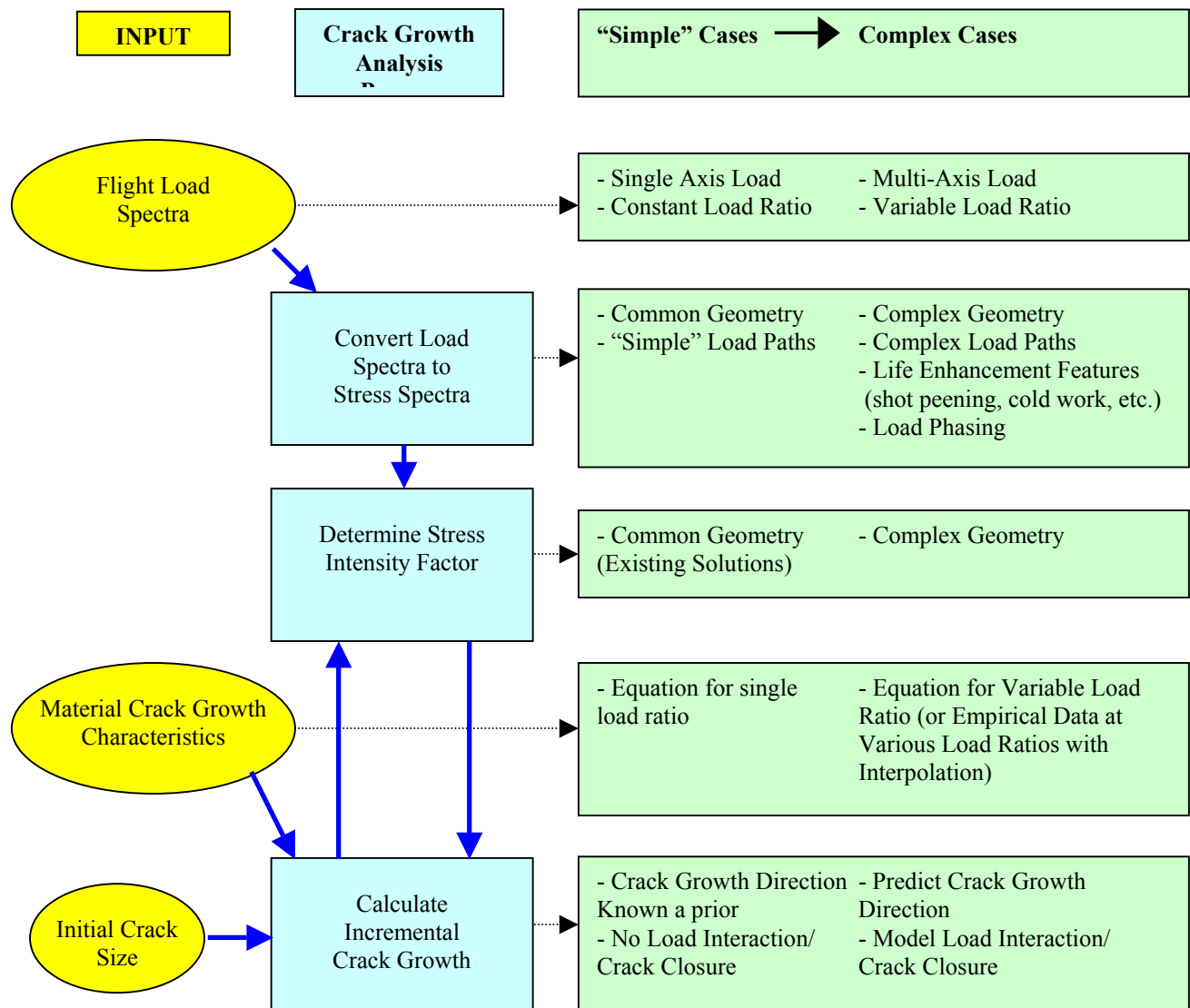
Figure 2-7 Details of Crack Growth Analysis Process.

An example of how sensitive rotorcraft crack growth can be to the choices made in analytically modeling the material crack growth properties is shown in Figure 2-8 through Figure 2-14. Five different NASGRO equation fits were developed for a material where da/dn vs. ΔK material characterization data was available for three different load ratios (R = -1, R = 0.01, R = 0.5). The NASGRO equation fits were developed by adjusting various equation parameters and comparing the results to the test data. The comparisons were made by generating the NASGRO equation curves for each of the test load ratios. The comparison for the original equation fit is shown in Figure 2-8. Four additional fits are compared in Figure 2-9 (fit 1), Figure 2-10 (fit 2), Figure 2-11 (fit 3), and Figure 2-12 (fit 4). All of these fits seem to be reasonable and it is difficult to select the "best" fit. Crack growth analyses were performed for each of the fits using the variable load ratio load spectra shown in Figure 2-13. The results are shown in Figure 2-14. The results show a significant variation in the analytical results due to the different NASGRO equation fits. The longest crack growth time (predicted using fit 4) exceeds the shortest crack growth time (predicted using the original fit) by more than a factor of 2.

One approach for demonstrating the suitability of a given choice for the analytical model used to represent material crack growth characteristics is to compare the analytical results to test data. Such an approach is demonstrated in section 3.3.

D357-T6 Aluminum 250 Deg F, Oil (R=-1, 0.01, 0.5) - Orig Fit
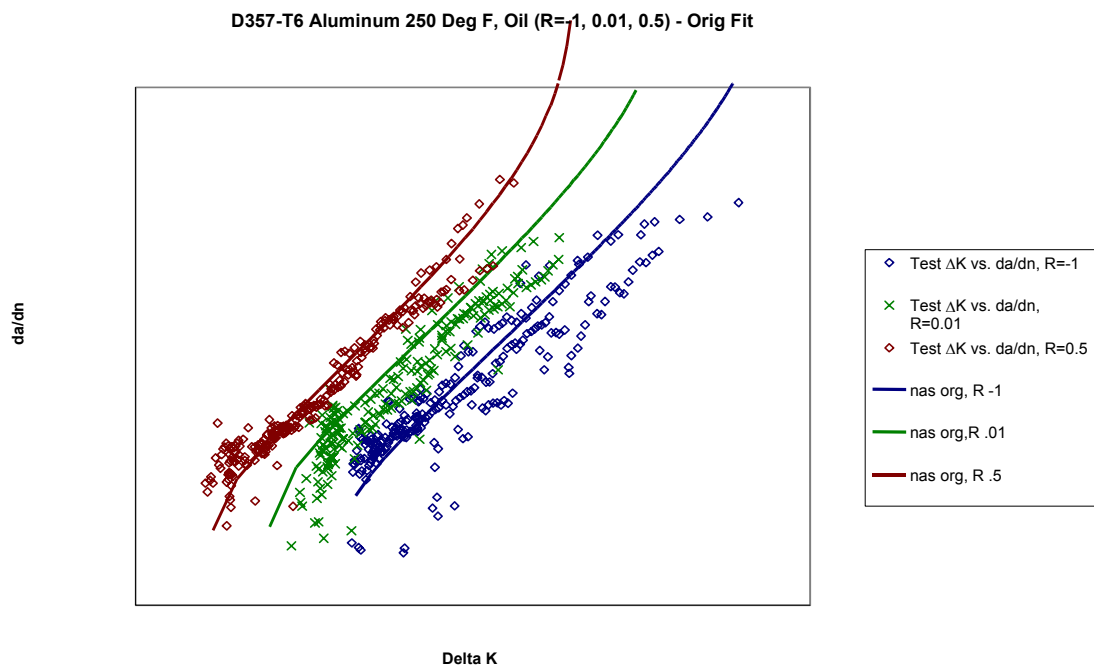
Figure 2-8 Original NASGRO equation fit to data, three load ratios.

D357-T6 Aluminum 250 Deg F, Oil (R=-1, 0.01, 0.5) - Fit 1
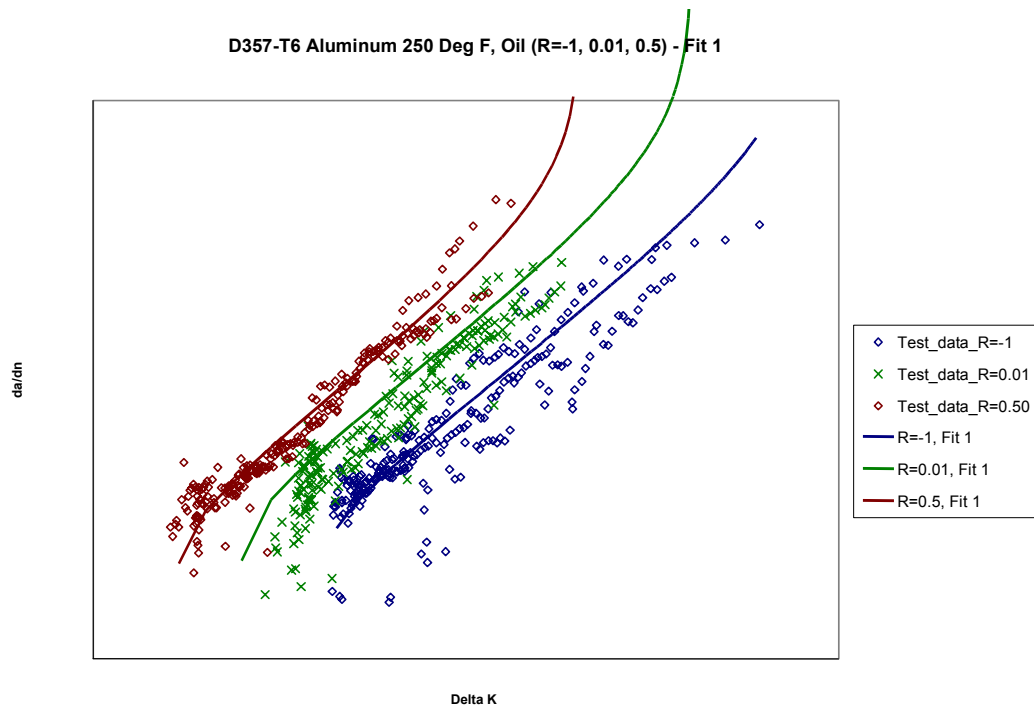
Figure 2-9 NASGRO equation fit 1 to data, three load ratios.

D357-T6 Aluminum 250 Deg F, Oil (R=-1, 0.01, 0.5) - Fit 2

Figure 2-10 NASGRO equation fit 2 to data, three load ratios.

D357-T6 Aluminum 250 Deg F, Oil (R=-1, 0.01, 0.5) - Fit 3
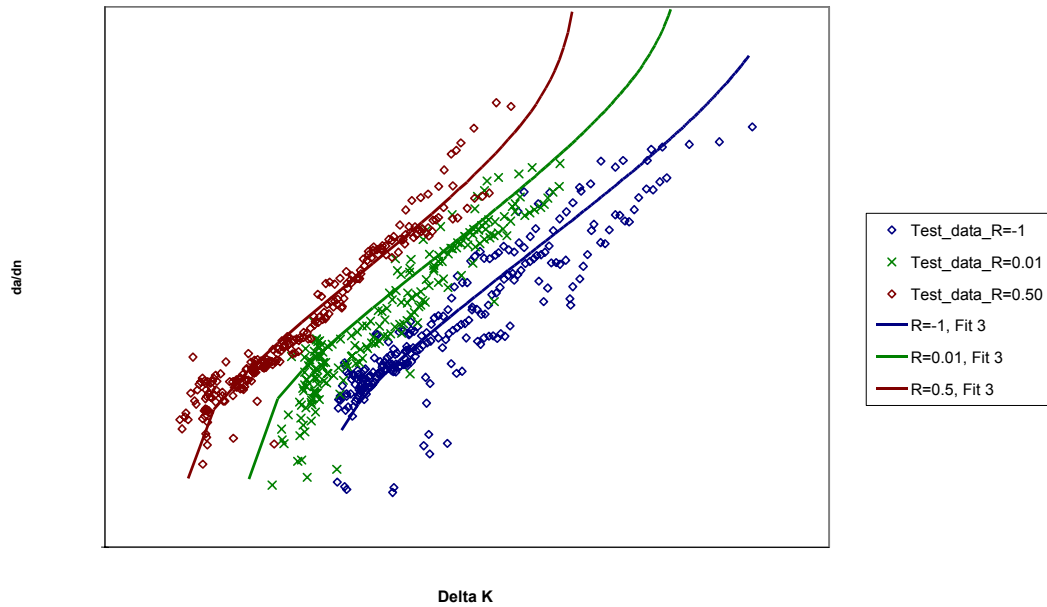
Figure 2-11 NASGRO equation fit 3 to data, three load ratios.

D357-T6 Aluminum 250 Deg F, Oil (R=-1, 0.01, 0.5) - Fit 4
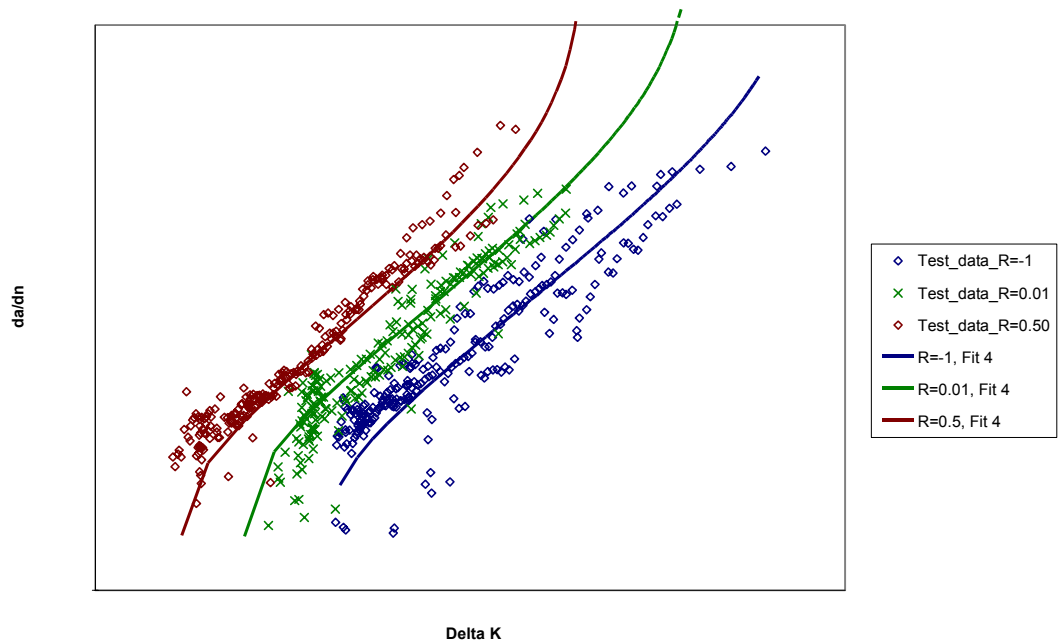
Figure 2-12 NASGRO equation fit 4 to data, three load ratios.
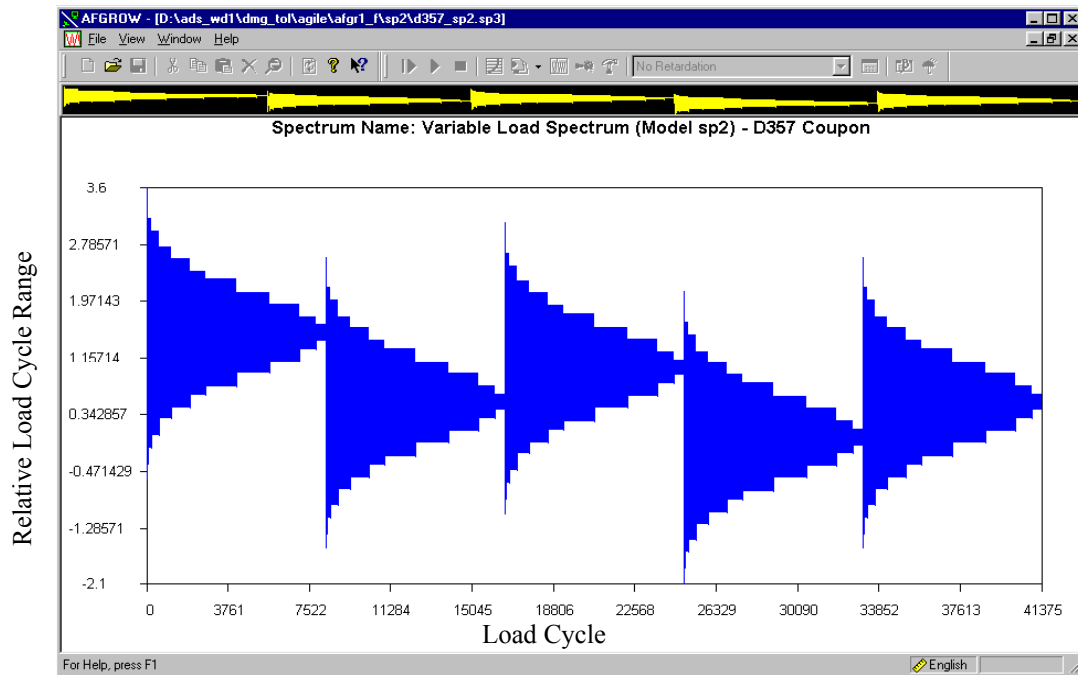
Figure 2-13 Variable load ratio spectrum used in study of data fit sensitivity.
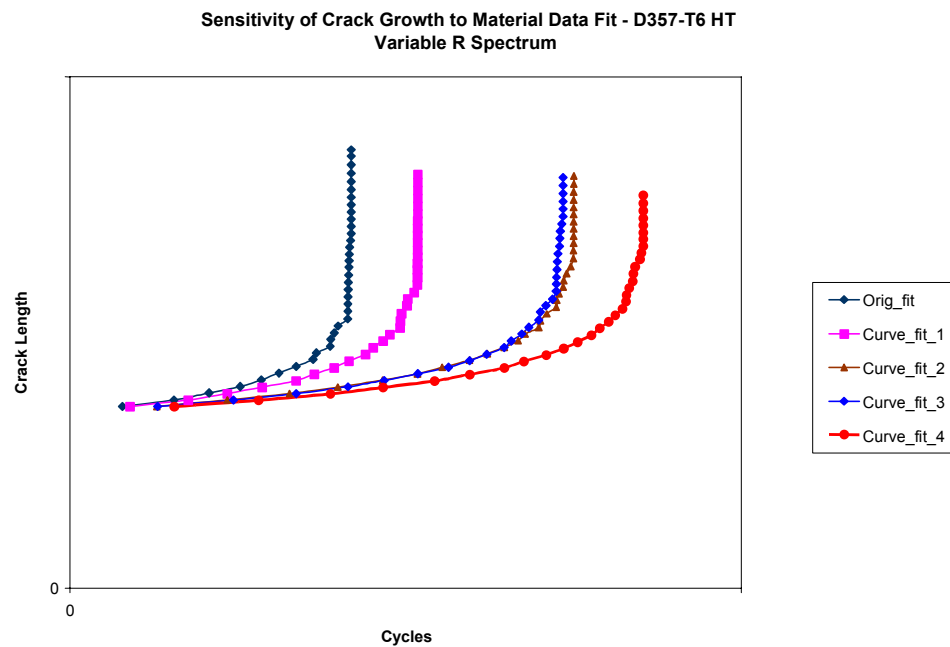
Figure 2-14 Comparison of analytical crack growth results for different fits to material crack growth characterization data.

2.7.2 Load Interaction Effects

Although the mathematical basis for the stress intensity factor is linear elasticity, and mathematically predicts a singularity in stress at the crack tip, in engineering materials a zone of plastic deformation is present in the region ahead of the crack tip. The size of the plastic zone is related to the magnitude of K_{max}. The extent to which a crack propagates for a given ΔK can be affected by the K_{max} from a previous load cycle. The effects of this plasticity can have significant implications for the calculation of crack growth under spectrum loading. The classic example of this is the case where the load sequencing includes a load cycle which includes a very high maximum load followed by a load cycle that reaches a much lower maximum load. Under linear elasticity, the second load cycle could result in a stress intensity factor range above the threshold, thus resulting in crack growth. Considering plasticity effects, the plastic zone created by the very high maximum load cycle results in residual stresses after the load is relaxed. During the second load cycle the rate of crack growth may be reduced as the crack grows through the plastic zone created by the higher stress level. Most crack growth analysis codes include various models that attempt to account for this behavior. Generally these mo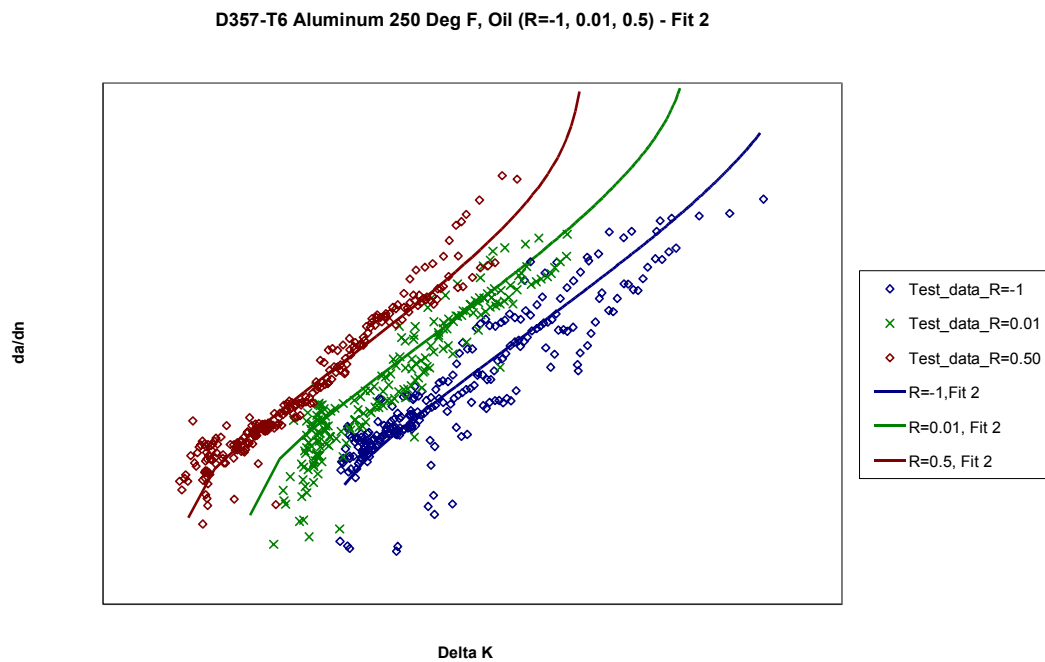dels reflect the effect of the high load plastic zone by reducing the effective stress ratio, R, for crack growth at the lower stress level. However, crack closure models reflect the effect of the plastic zone from the high stress by adjusting the crack opening stress at the lower stress level.

Due to the diverse nature of rotorcraft load spectra, the significance of load interactions effects can vary. An analytical example of this is presented in section 2.1.2. Crack growth analyses for the load spectrum from Figure 2-3 shows very little difference between an analysis that included modeling for load interaction effects versus analysis that did not include modeling for load interaction effects as can be seen in Figure 2-5. However, a similar comparison for the load spectrum from Figure 2-4 shows a significant difference as can be seen in Figure 2-6.

One of the challenges in crack growth analysis is to demonstrate that the load interaction modeling used is appropriate for a given load spectra sequencing (and material). An approach for addressing this issue is presented in section 3.3.

2.7.3 Stress Spectra and Stress Intensity Factor Determination

Discussions of crack growth analysis stress spectra and stress intensity factor determination related issues will be developed in a later version of this report.

2.7.4 Crack Growth Path

Discussions of crack growth analysis crack growth path related issues will be developed in a later version of this report.

2.7.5 Validation

Discussions of crack growth analysis validation related issues will be developed in a later version of this report.

2.7.6 Efficiency and Ease of Use

Discussions of crack growth analysis efficiency and ease of use related issues will be developed in a later version of this report.

2.8 Certification

Discussions of certification related issues will be developed in a later version of this report. See section 3 for descriptions of case studies and demonstrations that address some certification issues.

2.9 Inspection

Discussions of inspection related issues will be developed in a later version of this report.

2.10 Risk Assessment/Reliability

Discussions of risk assessment/reliability related issues will be developed in a later version of this report. See section 3 for descriptions of case studies and demonstrations that address some risk assessment/reliability issues.

2.11 Methodology and Component Management

Discussions of methodology and component management related issues will be developed in a later version of this report. See section 3 for descriptions of case studies and demonstrations that address some methodology issues.

3. Case Studies and Demonstrations

Bell, Boeing and Sikorsky (RITA companies) are investigating the specific issues related to the damage tolerance analysis methodology of rotorcraft by using Principal Structural Elements (PSE) for case studies and demonstration and validation of damage tolerance analysis (DTA) methodology. Bell Helicopter has selected a dynamic system component PSE, main rotor yoke of a medium lift helicopter for the research. Boeing Helicopter is investigating issues related to the certification and testing. Sikorsky Aircraft is investigating the issues related to the damage tolerance analysis of an airframe component PSE.

3.1 Dynamic System Component PSE Case Study (Bell)

Bell selected a dynamic system component PSE as a case study to demonstrate the damage tolerance analysis (DTA) methodology for surface cracking in structures with high cycle fatigue loading. The PSE Bell selected was a titanium main rotor yoke of a medium lift helicopter (see Figure 3-1). A flow diagram of the damage tolerance analysis methodology used in the RCDT program is shown in Figure 2-1. The specific issues study addresses the need to identify issues and define guidelines for implementation of damage tolerant design of rotorcraft structures in compliance with FAA guidance material. Some specific issues for damage tolerance (DT) design of rotorcraft structures are: crack growth material threshold data for dynamic components under high cycle fatigue loading, spectrum loading effects, validation of crack growth analysis methods for design and certification of rotorcraft components, practical field inspection methods, and application of probabilistic methods for risk assessment of DT designs. The benefits of this demonstration of DTA methodology will be guidance material to industry for addressing damage tolerance analysis and design issues.

3.1.1 Approach

For rotorcraft structures, the DTA methodology needs to be demonstrated and validated using testing. This will be accomplished by using a building block approach of analyzing and testing the selected main rotor yoke PSE. The testing and DTA validation will focus on the critical area of the flapping flexure (see Figure 3-1). The crack growth life prediction is a complex process that involves several different variables: crack growth threshold allowables, threshold testing methods, cyclic fatigue stresses, stress ratio, geometry of the component, diagnostics methods for determining initial crack size, the effects of load interactions; the physical environment such as humidity, presence of chemicals and temperature etc. and the experience of the analyst. Issues relating to the environment are not investigated but the following issues are going to be researched:

- Damage tolerance analysis of the rotor and dynamic components under high cycle fatigue with surface cracks
- ΔK_{th} (Crack growth threshold) definition and data and testing methods
- Spectrum Loading issues such as spectrum loading effects on crack growth, spectrum truncation and cycle counting methods
- Comparison of DT vs. safe-life design limits
- Correlation of predicted crack growth life from commercially available crack growth analysis codes with the generated test data

Figure 3-2 shows the building block approach for generating the required test data. A large amount of data including spectra for analyzing and testing, crack growth threshold and da/dN data, element level test data and full scale component test data is needed for the DTA validation. The following sections are devoted to the development of required input data.

3.1.2 Spectrum Development for Testing

Figure 3-2 and Figure 3-3 show schematics for developing a load spectrum and a stress spectrum for testing the coupons (basic and element level), full-scale component testing and analyzing the PSE with CGA codes. Detailed explanation about the spectrum development can be found in section 2.4 of Reference 4.

Several different spectra including the data from Bell's Loads and Database as shown in Figure 3-3 will be investigated to develop a normalized spectra for testing and analysis.

3.1.3 Coupon Testing

The test data will be generated from coupons, element level testing and full-scale component testing. Coupon testing will be used to generate the basic crack threshold data needed as input for the crack growth analysis. The cracks are in general either surface cracks e.g. in dynamic components, through cracks e.g. in airframe components and corner cracks e.g. in the holes of fittings, lugs or dynamic components. The crack growth threshold and da/dN data will be developed for all three different shapes of the cracks by using three different coupon configurations. For surface cracks crack growth threshold and da/dN data Kb bar coupon will be used. CT (compact tension) coupon will be used for through thickness cracks and a square bar coupon will be used for corner cracks. The data from all these coupons will be investigated and recommendations will be made with regard to the use of the crack growth threshold and da/dN data for the crack growth analysis of a main rotor yoke PSE and element level coupons using CGA codes.

Flapping Flexure Area

Material: Titanium 6Al-4V Annealed
Type III, Comp. 'C', Plate, MiL-T9046
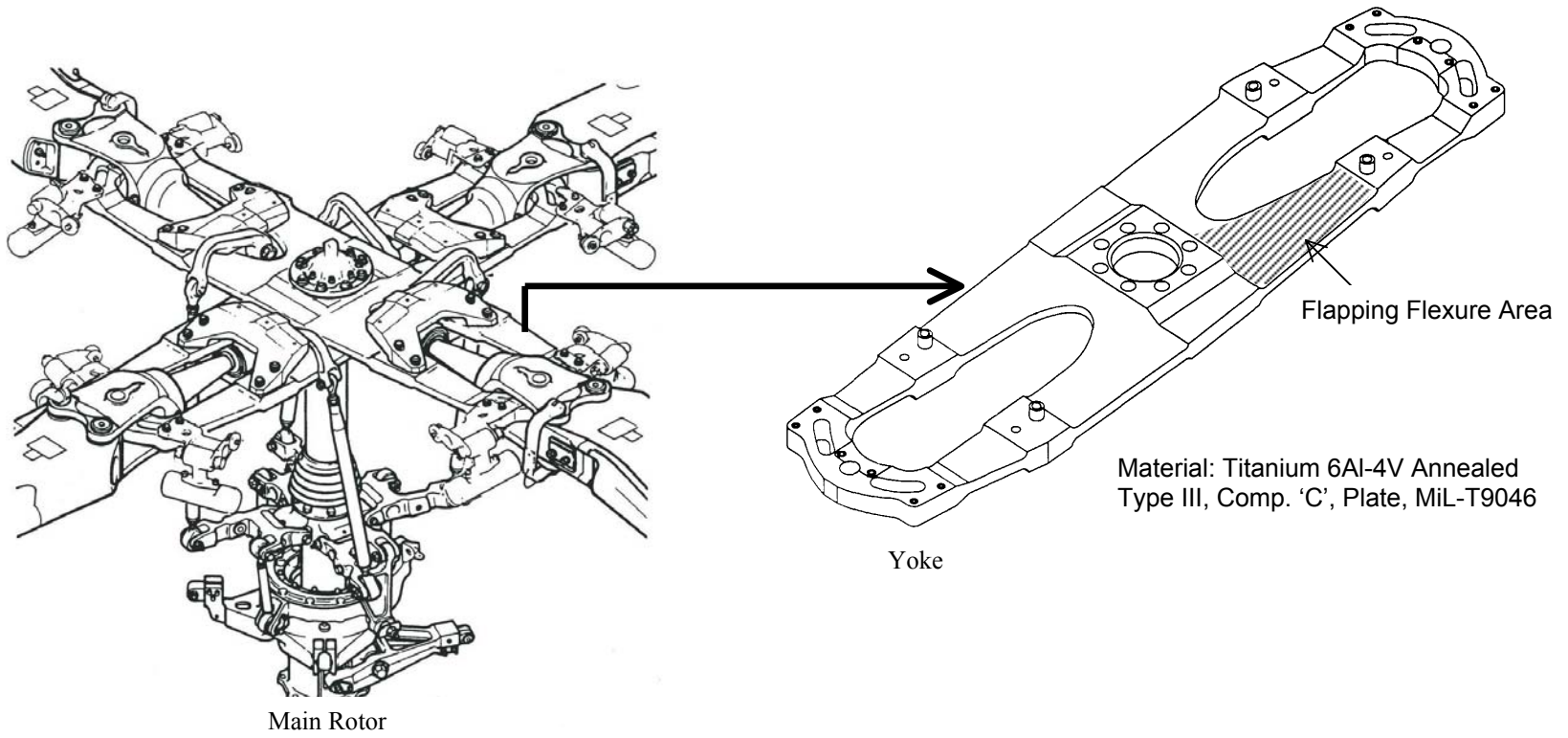
Yoke

Main Rotor

Figure 3-1 Main Rotor Yoke PSE.
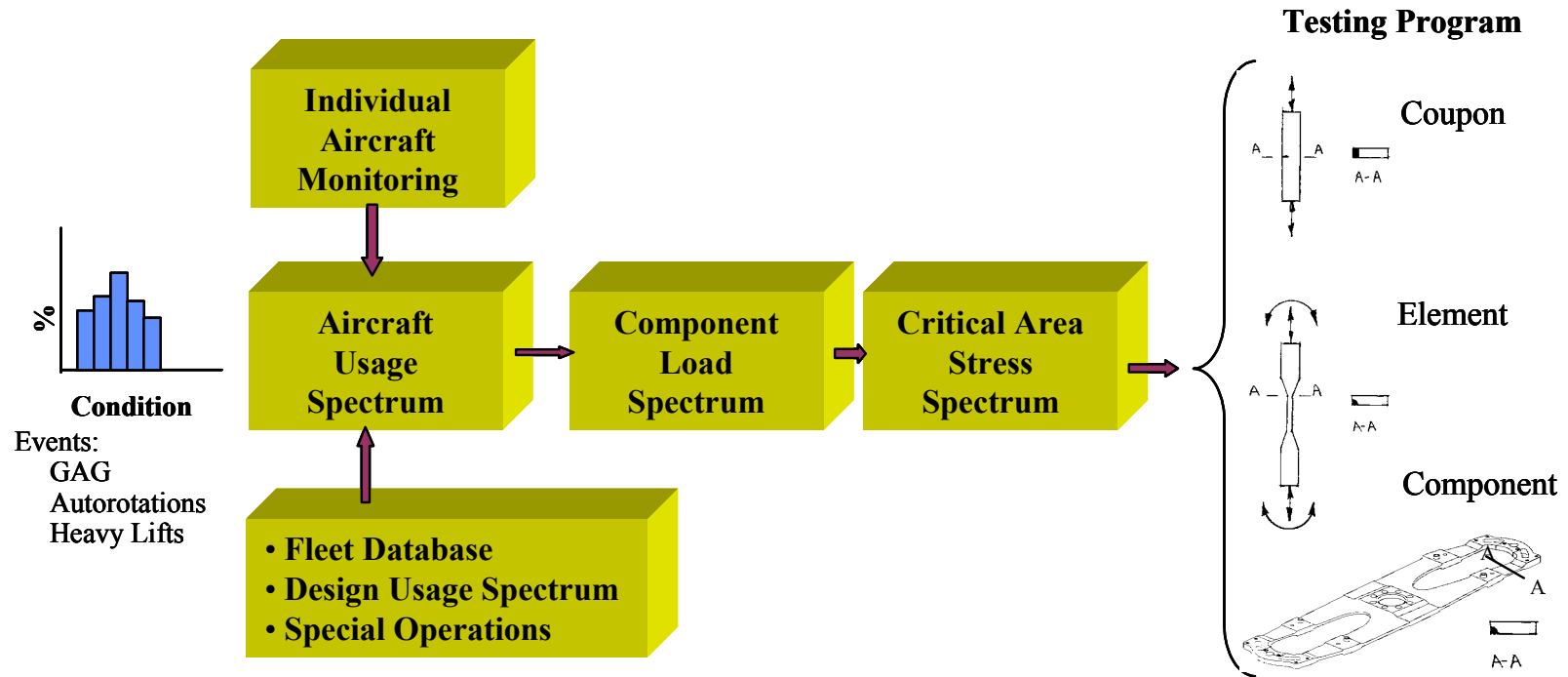
Figure 3-2 Building Block Approach of Demonstration of a DTA Methodology of a PSE.
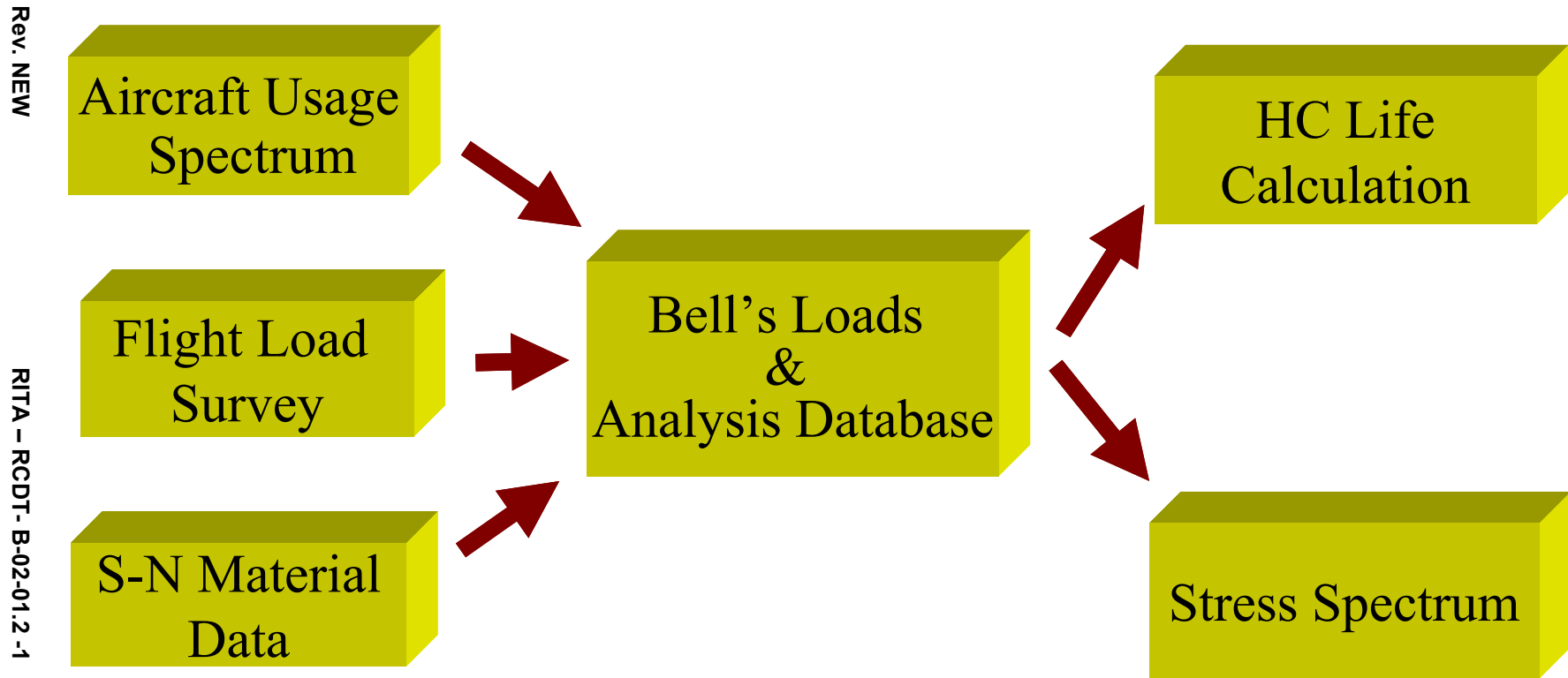
Figure 3-3 Stress Spectrum Development and High Cycle Fatigue Calculation from a Database.

3.1.4 Element Level Testing

The second level of crack growth testing will use element level specimen, which will replicate local geometry (e.g. thickness) and the primary loads of the PSE where the rogue crack is such that the local stress is same. The specimens will be tested using spectrum loads developed in section 3.1.2. Some of the specimens will be tested with and without surface treatments to evaluate the effects of the surface treatments. Also some specimens will be tested using different truncated spectra for determining the truncation of the loads that will be used for testing full-scale component. The data generated here in conjunction with basic coupon data will be used to validate the crack growth analysis codes by comparing the predictions with the test results. In addition the test data will also be used to address the issues and develop a test plan to test a full-scale component from spectrum load effects, surface treatment effects and truncation levels.

3.1.5 Full-Scale Component Testing

The third level of crack growth testing will be for a full-scale PSE component or subcomponent. Depending upon budget, a sub component representing the structure around the rogue crack may be made such that the loads and the geometry are comparable to the component. The specimen will be tested using the spectrum loads generated in section 3.1.2.

3.1.6 Crack Growth Life Prediction Using CGA Codes

Inputs for crack growth analysis consists of component geometry, crack growth threshold and crack growth da/dN data, a load spectrum and an initial crack size. For simple component geometry and loads the analyst can use CGA codes like AFGROW and NASGRO that have built in library of crack growth models. BEASY, AGILE and FRANC3D CGA codes can be used for complex geometry and loadings. The analysis usually consists of determining the stress intensity factors and determining the crack growth life for a given crack growth model and a load spectrum. CGA codes e.g. AFGROW, NASGRO and AGILE will be used for predicting the crack growth lives of element level coupons and full-scale component for a given geometry, loads and crack growth threshold and da/dN data developed in section 3.1.3.

3.1.7 Correlation of Predicted Life with Test Data

The data generated under this effort will help validate the building block methodology and will also provide data for validation of crack growth analysis codes such as AFGROW, NASGRO, and AGILE. In addition, the data will also

be used to demonstrate and validate if element level testing would suffice instead of conducting expensive full-scale testing for certification/qualification.

3.1.8 Risk Assessment

The Risk Assessment/Probabilistic Methods will be used to address the issues regarding uncertainties in material properties, crack sizes, loads, and inspection and the effect on the crack growth analysis and probability of failure.

3.2 Airframe Joint Damage Tolerance (Sikorsky)

This section will be added in a later version of this report.

3.3 Boeing Validation/Certification Testing

A fundamental capability to quantify or empirically characterize the basic ability to predict crack growth in rotorcraft components is required for the successful implementation of a comprehensive RCDT methodology. While crack growth based damage tolerance has many variables and uncertainties to consider, without an appreciation to what extent the basic capability exists to predict crack growth under known conditions, it is difficult to assess the risks associated with the many variables and uncertainties. For example, Figure 3-4 shows a representation of a core crack growth analysis. The basic given inputs are the component geometry, the crack growth properties for the component material, a load spectrum, and an initial flaw size. The core crack growth analysis consists of determining the stress intensity factor, determining the stress spectra, an analytical representation of the material crack growth properties, and modeling of load interaction effects. The assessment of the crack growth prediction capability is simply a comparison of the predicted results to the scatter from the results of multiple tests.

Ideally, the validation of crack growth analyses would be accomplished through full scale component testing. Historically, safe life (crack initiation) fatigue methodologies used in the rotorcraft industry have relied heavily on full scale component fatigue testing. For these safe life applications, the complex geometry, complex internal and external load paths, material processing effects, and size effects have minimized the ability to accomplish an acceptable capability to predict crack initiation based on analysis and coupon test data.

Unfortunately, full scale component crack growth testing is much more complex and time consuming, and consequently much more expensive, than crack initiation testing. For crack growth based damage tolerance the presumption of the existence of a crack most likely will mitigate uncertainties associated with size effects. In addition, the sophistication of analytical stress analysis tools has greatly increased resulting in an improved capability to handle complex geometry and loading. As a result, it is reasonable to attempt to develop RCDT methodologies that rely more on less expensive coupon or sub-element testing and less on more expensive full scale component testing, i.e. a building block based methodology.

An effective way to address the RCDT core methodology issues and to lay the ground work for a building block based methodology is to conduct research that initially addresses fundamental issues using coupon testing that can be followed up with additional testing of sub-elements and/or full scale components. Specifically, as shown in section 2.7.1, the crack growth analysis results can be sensitive to small variations in the way the material crack growth properties are represented analytically. In addition, as is shown in section 2.1.2, the significance of load interaction effects can vary between diverse rotorcraft load spectra. For a

given material, a well planned coupon testing program can provide a basis for establishing a combination of an analytical representation of the crack growth material properties for that material and a load interaction model that is applicable for a wide rotorcraft load spectra. If successful, this combination could be applicable to a wide range of components made from the same material. In addition to establishing the analytical representation of the material crack growth properties and the load interaction model, coupon testing is a convenient way to investigate the inherent scatter in crack growth by providing a relatively inexpensive means of performing a large number of repeat tests. This coupon testing would develop basic data and would fit into an overall building block approach as shown in Figure 3-5.

After coupon testing, the next step in the building block approach would be to conduct sub-element or full-scale component testing to verify that the coupon test results are applicable to more complex geometry and/or load paths. Initially, some of the sub-element or full scale component testing might require multiple tests to demonstrate the level of inherent scatter relative to the coupon data. Additional components with similar geometry and/or load paths would require limited testing. Over time, as a database of experimental data is built up for a given material, there is a potential that testing requirements could decrease.

GIVEN:		CORE CRACK GROWTH ANALYSIS:		VALIDATION:
Component Geometry Component Material Crack Growth Characterization Data (da/dn vs. ΔK and ΔK_{TH} for multiple load ratios) Load Spectrum Initial Flaw Size Environment	→	Stress Intensity Factor and Stress Spectra Calculations Analytical Representation of Material Crack Growth Properties (Equation or empirical) Modeling of Load Interaction Effects	→	Comparison of Predicted Crack Growth to the Scatter from Multiple Tests

Figure 3-4 Core Crack Growth Analysis.

EXTENSIVE TESTING

COUPON TESTING – Material A

Test Variables
- Diverse Rotorcraft Load Spectra;
- Range of Scale Factors
- Multiple Repeats at Each Scale Factor

Analysis Variables
- Analytical Modeling of da/dn vs. ΔK, ΔK_{TH}
- Load Interaction Models

Results
- Measure of Inherent Scatter (Risk Assessment Applications)
- Combination of Analytical Modeling of da/dn vs. ΔK, ΔK_{TH} and Load Interaction Model that Results in Predicted Crack Growth that is Consistent with Test Results for Different Load Spectra

LIMITED TESTING

SUB-ELEMENT/COMPONENT TESTING – Material A

Analysis Variables
- Analytical Modeling of da/dn vs. ΔK, ΔK_{TH} from Coupon Testing
- Load Interaction Model from Coupon Testing

Test Variables
- Geometry/Load Path Combination 1

Test Variables
- Geometry/Load Path Combination 2

Test Variables
- Geometry/Load Path Combination "n"

Results
- Measure of Inherent Scatter (Risk Assessment Applications)
- Validate Coupon Test Modeling for da/dn vs. ΔK, ΔK_{TH} and Load Interaction

APPLICATION TO SIMILAR COMPONENT/LOADING CONDITIONS – Material A

- **MINIMAL TESTING BASED ON SIMILARITY**
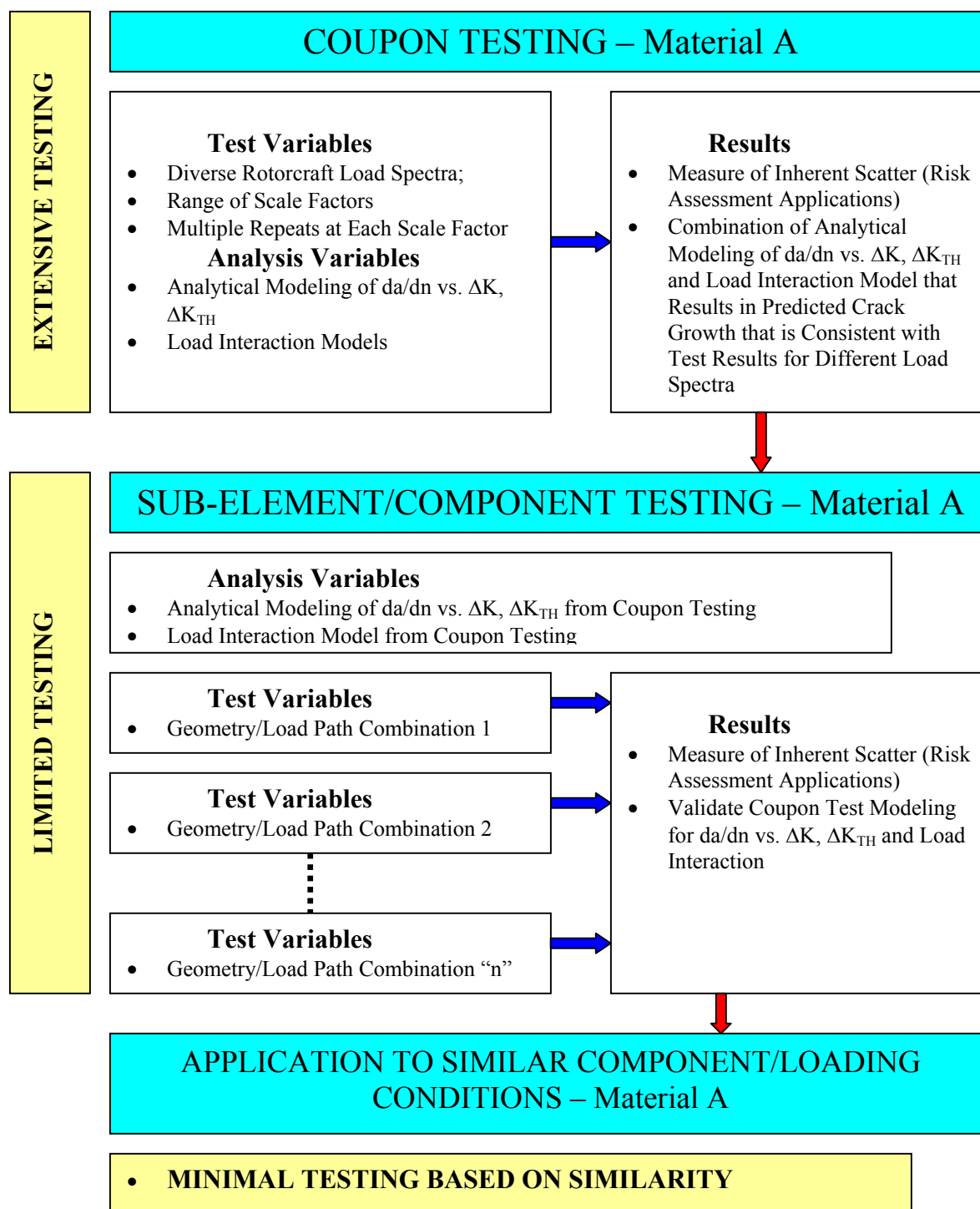
Figure 3-5 Building block approach to validating crack growth analysis.

3.3.1 Description of Coupon Testing

A coupon testing program is currently underway under the Boeing project. The program is focusing on two diverse load spectra similar to those shown in Figure 2-3 (based on reversed bending) and Figure 2-4 (based on drive torque). Materials selected for the program include several that are used in rotor components (forged 7050-T7452 aluminum, forged 6Al 4V beta annealed titanium, and forged 4340 steel), one material that is typically used in drive shafting components (9310 steel), and one gear steel (VASCO X2M). The specimens used in the testing are standard compact tension specimens per the specifications of ASTM E647.

3.3.2 Results of Coupon Testing

At the present time (October 2003), data is available for the results of testing of the forged 7050-T7452 aluminum using the torque based load spectra. A sample of the results is presented in Figure 3-6. The figure shows the results of four coupon tests where the scale factor applied to the normalized load spectra were the same (scale factor of 17). The scatter in the test data ranges from approximately the equivalent of 108 hours of flight to approximately 140 hours of flight.

Figure 3-6 also includes an analytical curve. This curve was developed after several iterations on the analytical representation of the crack growth material properties. Note that early in the crack growth the analytical curve tracks on the bottom of scatter of the test data. Later on the analytical curve moves into the center of the scatter. Further evaluation of the analytical models for crack growth properties and load interaction will be conducted when test data for the bending load based spectrum is available.

Comparison of Analytical Crack Growth and Test Data for Al 7050-T7452
(Torque Spectrum)
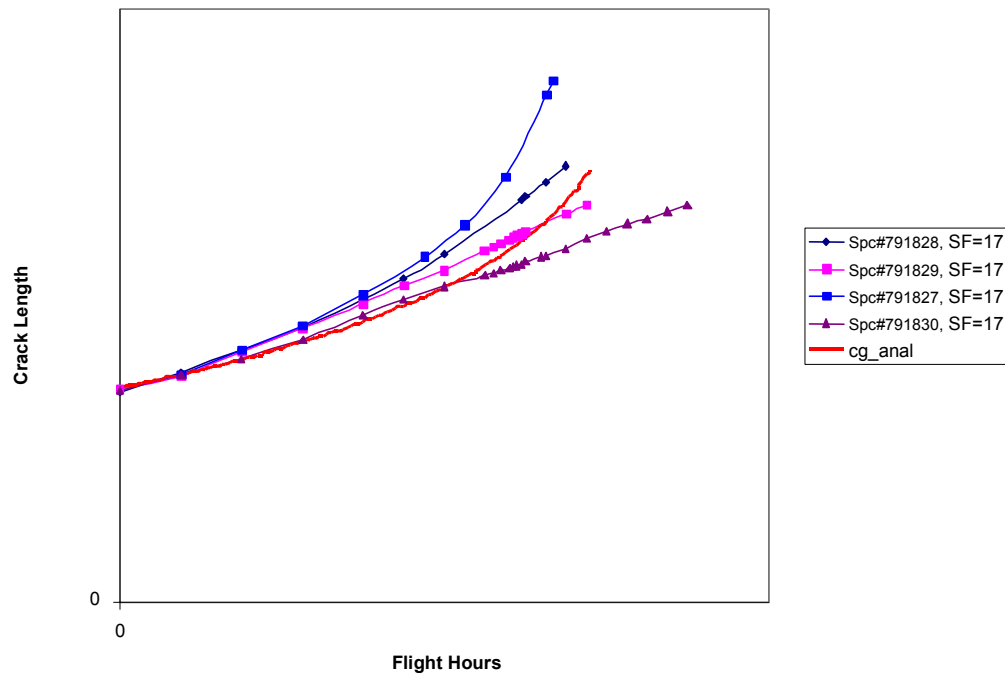
Figure 3-6 Sample results of load spectra coupon testing.

4. Recommendations

The application of fracture mechanics based damage tolerance methods to rotorcraft metallic structure includes many elements, each with varying levels of complexity. While it is important to develop an understanding of each of these elements, the full significance of the various features of each element can only be fully evaluated within the context of an overall methodology. This interim report makes a good start in identifying the key features of each element. Good progress has been made in identifying issues and approaches for addressing these issues within the context of an overall methodology. Specific recommendations regarding these issues will be made in the coming years when the results of the demonstrations of the specific rotorcraft components are available. It is recommended that this work be continued so that a stronger technical foundation is developed for the application of fracture mechanics based damage tolerance methods to rotorcraft metallic structure.

5. Glossary of Terms and Fracture Mechanics Basics

5.1 Glossary of Terms

Symbol or Term	Description
Alternating Load	The amplitude of a load cycle; equal to half the range from the maximum load in the cycle and the minimum load in the cycle. Also sometimes referred to as the Oscillating Load or the Vibratory Load.
CGA	Crack growth analysis
da/dn	Crack growth rate, i.e. incremental growth of crack length "a" per load cycle "n"
da/dn vs. ΔK	Crack growth rate as a function of stress intensity factor range; Dependant on material, environment, and load ratio.
DTA	Damage tolerance analysis
Flight regimes	The categories of steady state and transient maneuvers that cover all rotorcraft operational time
K	Stress intensity factor, a function of stress, geometry, and crack size
K_{MAX}	Stress intensity factor corresponding to the maximum stress for a vibratory load cycle
K_{MIN}	Stress intensity factor corresponding to the minimum stress for a vibratory load cycle
ΔK	Stress intensity factor range corresponding to a vibratory load cycle range, that is $\Delta K = K_{MAX} - K_{MIN}$ where K_{MAX} is a function of the maximum stress and K_{MIN} is a function of the minimum stress. Note that for negative load ratios, that is the minimum load is compressive (negative) and the maximum load is tension (positive) the stress intensity factor range has been defined in different ways in the literature. In this report, the convention whereby the actual signed value of the compressive minimum stress is used to calculate K_{MIN} is referred to as the "full range" definition of ΔK. The definition used in the ASTM specification E-647, as defined below, is referred to as the ASTM definition or ΔK_{ASTM}. It is important that the convention used in developing the da/dn vs. ΔK material properties used in an analysis are consistent with the convention used by the analytical code when evaluating crack growth that involves negative load ratio loads.
ΔK_{ASTM}	Definition of the stress intensity factor range corresponding to a vibratory load cycle from ASTM E-647, where $\Delta K = K_{MAX} - K_{MIN}$ for load ratios greater than zero $\Delta K = K_{MAX}$ for load ratios less than or equal to zero
ΔK_{TH}	Threshold stress intensity factor range; for ΔK values below this threshold crack growth is presumed to not occur.
Load Ratio	The ratio of the minimum load to the maximum load in a load cycle.
Mean Load	The load value equal to the average of the maximum load and the minimum load in a load cycle. Also sometimes referred to as the Steady Load. The mean load plus the alternating load equals the maximum load for a load cycle. The mean load minus the alternating load equals the minimum load for a load cycle.
NDI	Nondestructive Inspection
NDE	Nondestructive Evaluation

Symbol or Term	Description
Oscillating Load	The amplitude of a load cycle; equal to half the range from the maximum load in the cycle to the minimum load in the cycle. Also sometimes referred to as the Alternating Load or the Vibratory Load.
Profile	A quantification of rotorcraft usage in terms of occurrence rates (typically number per hour or percent time per hour) of flight conditions (steady state maneuvers, transient maneuvers, and low cycle loading such as ground air ground cycles). Can also include splits in terms of percent time for aircraft configuration variables (gross weight, center of gravity, cargo type – external or internal) or other categories such as altitude. Typically expressed in terms of a matrix with rows of flight conditions and columns of splits on aircraft configuration or other categories.
PSE	Principal Structural Element
RCDT	Rotorcraft Damage Tolerance
Steady Load	The load value equal to the average of the maximum load and the minimum load in a load cycle. Also sometimes referred to as the Mean Load. The steady load plus the alternating load equals the maximum load for a load cycle. The mean load minus the alternating load equals the minimum load for a load cycle.
Stress Ratio	The ratio of the minimum stress to the maximum stress in a stress cycle.
Usage	The occurrences and sequences of flight regimes for rotorcraft operational time. See Profile
Vibratory Load	The amplitude of a load cycle; equal to half the range from the maximum load in the cycle and the minimum load in the cycle. Also sometimes referred to as the Alternating Load or the Oscillating Load.

5.2 Fracture Mechanics Basics

The predominant approach in predicting crack growth in engineering applications makes use of the linear elastic fracture mechanics stress intensity factor parameter. The stress intensity factor, K, is a mathematical parameter used in the expressions that define the stress fields in the vicinity of a crack tip singularity. The stress intensity factor is a function of component geometry, crack geometry, and a reference stress (e.g. a far field stress). The stress intensity factor can be expressed as:

$$K = f(g) * \sqrt{a} * \sigma$$

where f(g) represents the dependency on component and crack geometry, "a" represents the crack length (or half length), and σ represents a reference stress such as a far field stress.

In linear elastic fracture mechanics, the mathematical equations that describe the stresses and displacements in the vicinity of a crack in an arbitrarily loaded elastic body are developed in terms of the relative displacements of the crack surfaces. Three such displacement modes are needed to define the possible

relative displacement of the two surfaces of a crack. A stress intensity factor is associated with each mode.

The three modes are shown in Figure 5-1. Mode I represents the case where the forces oriented normal to the crack surface pull the crack surfaces apart in a direction normal to the crack surfaces. The direction of the crack surface displacements are also normal to the direction of crack growth. This is referred to as the crack opening mode and is of primary interest for most engineering applications. The Mode I stress intensity factor is expressed as K_I. Due to its dominance in engineering applications, the subscript is often not used. In this report, any reference to stress intensity factor that does not include a reference to the mode should be assumed to be Mode I.

Mode II represents the case where forces oriented parallel to the crack surfaces and parallel to the direction of crack growth cause the crack surfaces to slide across one another in the same direction as the direction of crack growth. This is referred to as the shear mode. The Mode II stress intensity factor is expressed as K_{II}.

Mode III represents the case where forces oriented parallel to the crack surface and normal to the direction of crack growth cause the crack surfaces to slide across one another in a direction perpendicular to the direction of crack growth. This is referred to as the tearing mode. The Mode III stress intensity factor is expressed as K_{III}.

The stress intensity factor is not dependent on material. However, important material fracture mechanics properties are defined in terms of the stress intensity factor. These material properties are independent of component geometry (except for fracture toughness under plane stress as discussed below). Under static loading, for a given material the critical stress intensity factor, K_C, referred to as the fracture toughness, identifies the stress intensity factor magnitude (for whatever combination of component geometry, crack length, and applied stress that result in that magnitude) at which static fracture of the component will occur. For cracks under plane strain conditions, the fracture toughness, or critical stress intensity factor, for a given material is a unique material property and is identified as K_{IC} (mode I). Under conditions where plane stress conditions exist to the extent that significant plastic deformation can occur during fracture, the actual fracture toughness can be numerically larger than the plane strain fracture toughness. This plane stress fracture toughness does vary with component geometry (for instance thickness) to the extent that these geometry variations affect the development of plastic deformation during fracture.

Under fatigue loading the stress intensity factor range, ΔK, is used to characterize a fatigue load cycle. For a given load cycle consisting of a maximum stress, σ_{max}, and a minimum stress, σ_{min}, the stress intensity factor range is

described in terms of a corresponding maximum stress intensity factor and a minimum stress intensity factor. That is, for

$$K_{max} = f(g) * \sqrt{a} * \sigma_{max} \qquad \text{and}$$

$$K_{min} = f(g) * \sqrt{a} * \sigma_{min}$$

$$\Delta K = K_{max} - K_{min} .$$

For a given material, the crack growth under fatigue loading is expressed in terms of the relationship between ΔK and the corresponding crack growth per load cycle, da/dN, that is developed empirically from crack growth rate testing. This da/dN versus ΔK relationship for a given material varies with load ratio R (where R = $\sigma_{min}/\sigma_{max}$) and other variables such as environment. It is important to note that for negative load ratios (that is σ_{min} compressive or less than zero) there are two definitions of ΔK that are used. One definition that is commonly used to document material crack growth properties and that is used in many crack growth calculation codes simply follows the above equations for all load ratios. A second definition set forth in ASTM specification E647-99 sets K_{min} to zero (that is $\Delta K = K_{max}$) for all negative load ratios. When performing crack growth analyses that involve fatigue load spectra that include negative load ratio load cycles, the convention for defining ΔK used in developing the material crack growth characteristics must be consistent with the convention used for defining ΔK in the crack growth analysis code.

The da/dN vs. ΔK relationship for a given material and load ratio is established by test. The relationship typically features three regions of crack growth behavior as shown in Figure 5-2. Region I crack growth refers to a low ΔK range where there is a transition from no crack growth to very small rates of crack growth (referred to as Region I crack growth). The ΔK value at which this transition occurs is called the threshold stress intensity factor range, ΔK_{TH}. Region II crack growth occurs at higher values of ΔK and is characterized by stable crack growth where the relationship between da/dN and ΔK often tends to exhibit log-log linear behavior. Region III crack growth occurs as the maximum stress intensity factor approaches the fracture toughness of the material and is characterized by a rapid increase in da/dN relative to increases in ΔK approaching an asymptotic value at the ΔK at which K_{max} equals the fracture toughness.

Although the mathematical basis for the stress intensity factor is linear elasticity, and mathematically predicts a singularity in stress at the crack tip, in engineering materials a zone of plastic deformation is present in the region ahead of the crack tip. The effects of this plasticity can have significant implications for both the experimental determination of material crack growth characteristics and for the calculation of crack growth under spectrum loading. The plasticity effects that can influence the experimental determination of the crack growth threshold stress

intensity factor range are of critical importance for rotorcraft applications. Due to the high rate at which load cycles can accumulate in rotorcraft, small errors in the representation of the crack growth threshold can have a large effect on the predicted crack growth time. Procedures for selecting load levels for this crack initiation step such that the resulting plastic deformation would have a negligible effect on the resulting threshold measurement have been set forth in ASTM specification E647-99. However, recent research suggests that in some cases these procedures can result in unconservative values for the threshold stress intensity factor. Efforts to clarify these results and to develop other experimental processes are currently (October 2003) being pursued under other government-funded research.

Plasticity effects can also influence crack growth under spectrum loading. The classic example of this is the case where the load sequencing includes a load cycle which includes a very high maximum load followed by a load cycle that reaches a much lower maximum load. Under linear elasticity, the second load cycle could result in a stress intensity factor range above the threshold, thus resulting in crack growth. Considering plasticity effects, the plastic zone created by the very high maximum load cycle results in residual stresses after the load is relaxed. During the second load cycle the rate of crack growth may be reduced as the crack grows through the plastic zone created by the higher stress level. Most crack growth analysis codes include various models that attempt to account for this behavior. Generally these models reflect the effect of the high load plastic zone by reducing the effective stress ratio, R, for crack growth at the lower stress level. However, crack closure models reflect the effect of the plastic zone from the high stress by adjusting the crack opening stress at the lower stress level.

Mode I (Opening Mode)

Crack Surface Displacements Perpendicular to
Crack Surfaces and Crack Direction

Crack Direction

Mode II (Shear Mode)

Crack Surface Displacements Parallel to Crack
Surfaces and Crack Direction

Mode III (Tearing Mode)

Crack Surface Displacements Parallel to Crack
Surfaces and Perpendicular to Crack Direction

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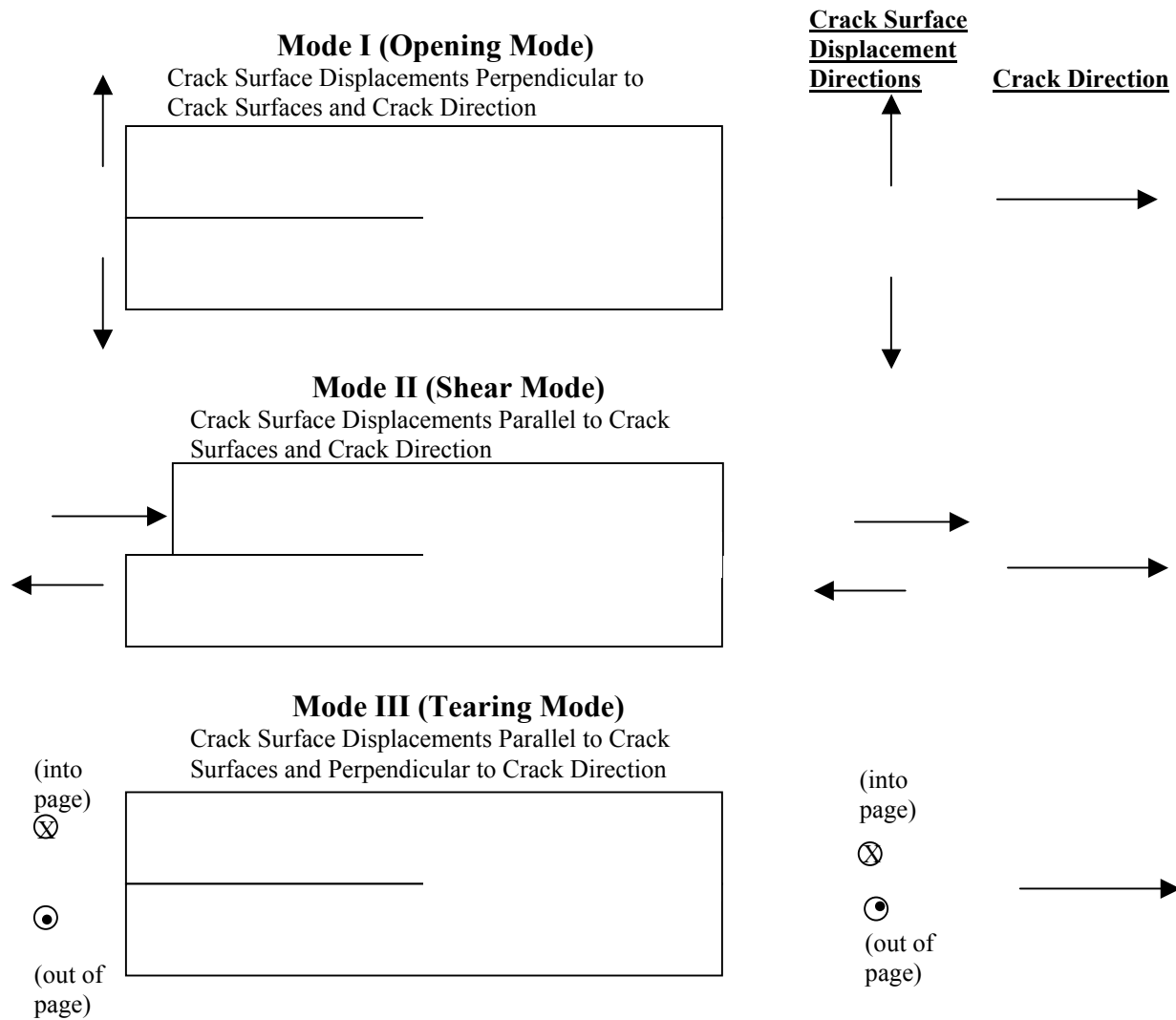(out of
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Figure 5-1 The three modes of displacement of crack surfaces relative to crack direction.
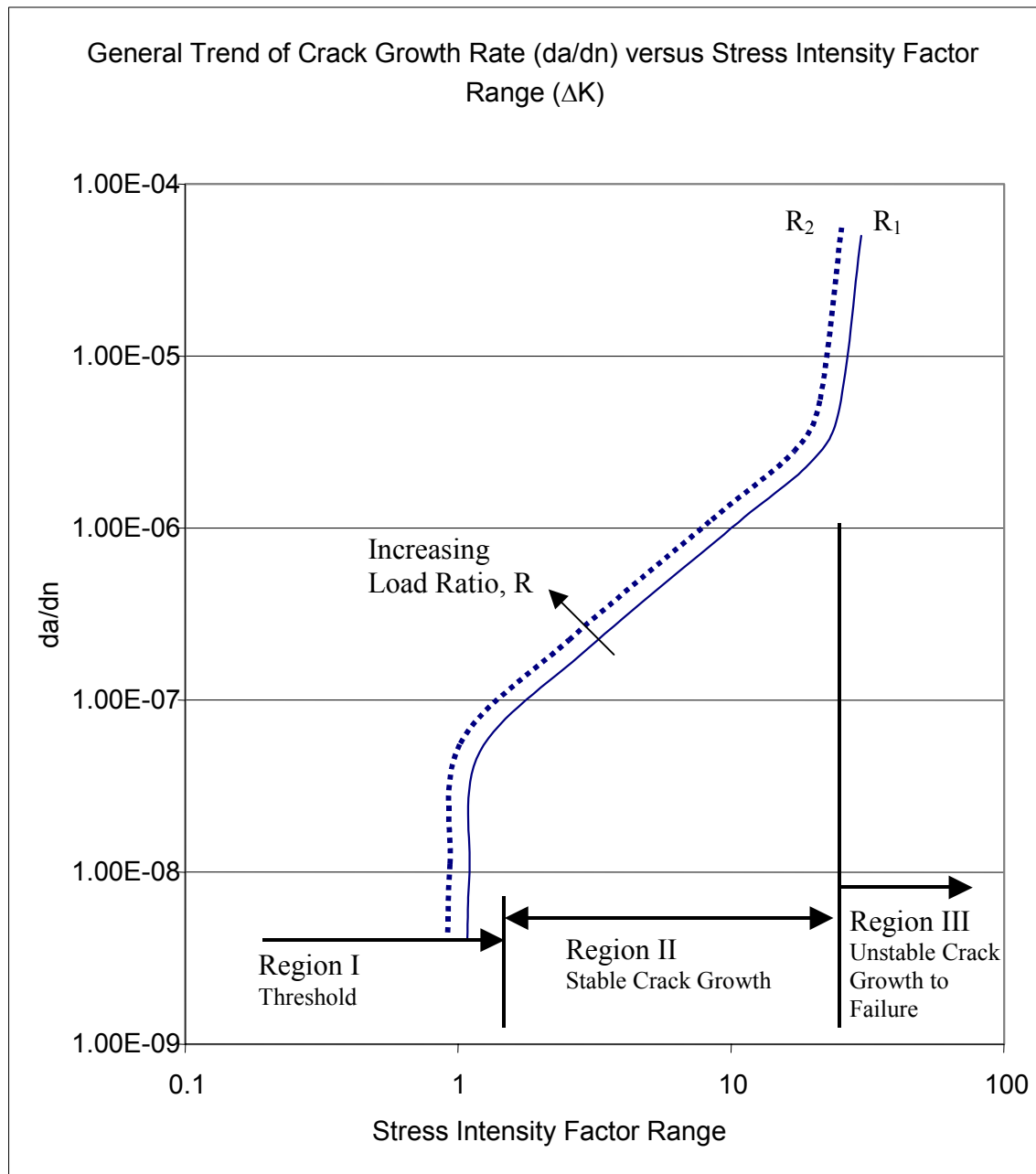
Figure 5-2 Typical trend in crack growth rate (da/dn) versus stress intensity factor range (ΔK).

6. References

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